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The Solar Powered Low-Acceleration Transport

After the turn of the century, the United States plans to have a manned base on Mars. The inhabitants of this base will wish to aerially explore the surrounding planet. For this purpose, a manned Mars aircraft will be required. A spacecraft will be required to deliver this aircraft to Mars. The purpose of this presentation is to present a conceptual design for this spacecraft system.

The aircraft design submitted to us by our aircraft designers has the rather unique property that the lift on the wings is insufficient to lift the weight of the plane. While this property is not desirable, it has been inspirational. Hence, we give you:

SPLAT

Solar Powered Low Acceleration Transport

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The Solar Powered Low-Acceleration Transport

The project objectives and requirements are as follows:

1) Develop a conceptual design for the spacecraft system required to deliver a manned aircraft to Mars.

2) The spacecraft will consist of a payload re-entry system and a science satellite with instruments for sensing the Martian surface. The satellite will remain in orbit after separating from the re-entry module.

3) The spacecraft's components and payload will be delivered to Earth orbit by the space shuttle and assembled at the space station. The extent of shuttle support should be identified and minimized.

4) The spacecraft will be able to be retrieved by a remote manipulation device on the space station or shuttle.

5) Nothing in the spacecraft's design should prevent it from performing many different missions of similar nature.

6) The spacecraft will have a design lifetime of at least four years.

7) The vehicle will use the latest advances in artificial intelligence where applicable to enhance reliability and lower costs.

8) The design will stress simplicity, reliability, low mass, and low cost.

9) For planning and costing, it will be assumed that four vehicles will be built.

10) The spacecraft will be able to perform the mission science objects.

The design process has been broken down into seven subsystems:

1) Mission Management, Planning, and Costing

2) Structure and Thermal Control

3) Aerobraking and Re-entry

4) Attitude and Articulation Control (AACS)

5) Science Instrumentation

6) Power and Propulsion

7) Command and Data Control

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Introduction:

The purpose of this section is to plan the mission and to calculate the cost of building the spacecraft. The important results provided in this section are as follows:

1) The ΔV required to achieve the escape from Earth and embark on the orbital transfer

2) The hyperbolic approach speed to Mars

3) The ΔV required to deorbit the reentry system from the satellite orbit

4) The launch windows

5) The cost of the mission

6) And the timetable for implementing the spacecraft design

The applicable Request For Proposal (RFP) requirements are:

1) Develop a conceptual design for the spacecraft system required to deliver a manned aircraft to the Martian surface in the first decade of the 21st century.

2) The spacecraft must consist of a payload reentry system and an instrument bus carrying scientific instruments for remote sensing of the planet's surface. The instrument bus must remain in orbit after separation from the reentry system.

3) The spacecraft's components and payload will be assembled on orbit at (and launched from) the space station.

4) The design will stress, among other things, low mass.

5) For cost estimating and overall planning, it will be assumed that four space delivery systems will be built.

6) The spacecraft must perform the mission science objectives.

These requirements will be satisfied by the following means. The first step is to determine the targets which best satisfy the mission objectives. This involves selecting an orbit to place the science satellite in and a deorbit trajectory to land the payload on Mars. With these targets specified, it will then be possible to determine the means of reaching them. This will involve planning a solar orbit transfer trajectory from Earth

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to Mars. Once planned, an Earth escape trajectory which will place the spacecraft onto this transfer orbit can be computed. With this information, the other subsystem designers will compute the mass breakdown for each subsystem, and, using the results, the cost of the mission will be calculated. Finally, an implementation plan will be assembled from the information obtained from the other subsystem designers.

Satellite Orbit:

The fist step is to select the instrument bus orbit. Three orbits were considered: a surface stationary orbit, a low altitude equatorial orbit, and a low altitude polar orbit.

The surface stationary orbit radius comes from solving the equation $R^3 = \mu T^2/4\pi^2$, where $\mu = 4.292 \times 10^4\ km^3/s^2$ is the gravitational constant times the mass of the central body, T is the period of the orbit, and R is the radius of the orbit. A Martian day is 24 hours, 37 minutes, and 23 seconds long. Hence, T= 88,643 sec. and so R= 20,442 km. The next step is to make a quantitative evaluation of the properties of this orbit. A positive number indicates a good property and a negative number indicates a bad property.

Property:	Value:
The satellite will always be overhead to assist the airplane	+ 1
Bad instrument resolution	- 3
Can only view half the planet	- 2
Satellite orbit will be disrupted by the passing of Deimos	- 4
Total evaluation of this orbit	- 8

Note the low value placed on assistance to the airplane. This is due to the fact that the aircraft designers have not indicated that there is such a need. This is not a very desirable orbit.

The next orbit considered is a low altitude orbit with an orbit radius of 3,895 km or an altitude of 500 km. The period of this orbit, calculated using the equation in the previous paragraph, is just over two hours. The orbit is evaluated as follows:

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Property:	Value:
The satellite will sometimes be overhead to assist the airplane.	+0
Most of the planet can be viewed	+3
Good instrument resolution	+3
Undisturbed orbit	+0
Total evaluation	+6

This orbit has many desirable properties.

The third orbit considered is a polar orbit 500 feet above the surface. It would also have a period of two hours. The orbit is evaluated as follows:

Property:	Value:
All of the planet can be viewed	+5
Good instrument resolution	+3
Undisturbed orbit	+0
The satellite will seldom be overhead to assist the airplane	-1
Total evaluation	+7

Because this orbit has the highest evaluation score, it was selected for this mission.

Deorbit:

The next step is to calculate a deorbit trajectory for the reentry system. Because of it's low ΔV requirement, a Hohmann style deorbit was selected. The apolapse of this orbit is 500 km above the surface and the periapse is on the surface. After separating from the science satellite, the reentry vehicle will need to execute a propulsive burn to transfer from the circular science satellite orbit to this deorbit trajectory. For the circular orbit, the spacecraft's velocity is calculated from the equation $V^2=\mu/R$. This equation gives V=3,320 m/s. For an object at the apolapse of the deorbit trajectory, the velocity is calculated from $V^2=\mu(2/R-1/a)$, where a=3645 km is the size of the semi-major axis. This equation gives V=3,204 m/s. Hence the reentry vehicle must fire a propulsive burn to provide a ΔV of 116 m/s to enter the deorbit trajectory.

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Earth to Mars:

Now the trajectory taking the vehicle from the Earth's solar orbit to Mars's solar orbit must be considered. To simplify the analysis, it is assumed that only the Sun's gravity will affect the spacecraft. Again, because of its low ΔV requirement, a Hohmann transfer trajectory will be used.

We begin with the assumption that the spacecraft has escaped the Earth's influence and is travelling in the Earth's orbit. Its velocity around the Sun is calculated from the equation $V^2=\mu/R$, where $\mu=1.327\times10^{11}$ km^3/s^2 is the gravitational constant times the mass of the Sun and $R=1.496\times10^8$ km is the radius of the Earth's orbit. We find that V=29.78 km/s. Mars orbits at a radius of 2.279×10^8 km. Hence, a Hohmann transfer orbit would have a semi-major axis $a=1.888\times10^8$ km. The velocity at periapse for this orbit can be calculated from the equation $V^2=\mu(2/R-1/a)$. The periapse of the orbit is the Earth's orbit so the above equation gives V=32.73 km/s. Hence, to enter onto the Hohmann transfer, a ΔV=32.73-29.78=2.95 km/s is required. This ΔV is the necessary hyperbolic excess speed required when the spacecraft escapes Earth's gravity.

The Martian orbit corresponds to the apolapse of the Hohmann transfer. When the spacecraft reaches this point, it's velocity can once again be calculated from the equation $V^2=\mu(2/R-1/a)$, where R is now the radius of the Martian orbit. This gives V=21.48 km/s. The velocity of Mars is calculated from $V^2=\mu/R$. This equation gives V=24.15 km/s. From a Martian point of view, the spacecraft will be approaching Mars at a V=24.15-21.48=2.67 km/s. This approach velocity is used to make aerobraking calculations (see Aerobraking and Re-entry).

Earth Escape-SEP vs. Chemical:

The spacecraft will be launched from the space station, which will be located at an altitude of about 300 km. To propel the spacecraft away from the Earth, the choice

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was narrowed down to two alternatives (see Power and Propulsion): a chemical system (I_{sp}=300 s) and a Solar Electric Propulsion (SEP) system (I_{sp}=4000 s).

First, consider the chemical system. This system will propel the spacecraft hyperbolically away from the Earth and onto the elliptical Hohmann transfer. While there are more advanced tools, such as MULIMP, which take the influence of other gravitational bodies into consideration, there are no similar tools that can be used for analyzing the SEP system. It was for the purpose of making an objective comparison that this simplified analysis was used.

The hyperbolic excess speed required is V_∞=2.95 km/s. The velocity of the spacecraft in orbit about the Earth at 300 km, V_0, can be calculated from the relation $V^2=\mu/R$, where $\mu=3.982\times10^5$ km^3/s^2 and R=6675 km. This gives V_0=7.724 km/s. The ΔV required can be calculated from the relation:

$$\Delta V=\sqrt{V_\infty^2+2\mu/R}-V_0$$

which give ΔV=3.59 km/s for this case. The ratio of fueled to empty mass can be calculated from the equation $M_f/M_e=e^{(\Delta V/c)}$ where c is the exhast velocity and is equal to $g_0 I_{sp}$. This gives the mass ratio to be 3.391. The mass that must be placed onto the Hohmann transfer is 4,269 kg. Thus, the spacecraft will be 19,000 kg, including the 1340 kg of engine inerts.

Now consider the SEP system. While it has a very high I_{sp}, it only produces a thrust of 2N. Because of this, the spacecraft will escape Earth along a spiralling-out trajectory, making several hundred orbits in the process. Analyzing this system's performance is difficult because little research has been done on low-thrust trajectories. A numerical integration of the equations of motion wasn't possible with the computing power available so another numerical method, the method of Multiple Elliptic and Hyperbolic Trajectories, was developed for this design. Using it, the ΔV and

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the escape time required can be determined. This method is based on the following simplifying assumptions:

1) The spacecraft's trajectory while accelerating over some interval, Δt, will be the same as that of a spacecraft that is accelerated the same amount impulsively at the beginning of the time interval and then coasts for the duration.

2) The spacecraft's acceleration is constant over Δt.

3) The change in the spacecraft's orbit radius is insignificant compared to the orbit radius.

4) The spacecraft's thrust is always applied in the direction of the spacecraft's velocity vector.

5) The spacecraft doesn't drift significantly from the Earth's solar orbit during the burn.

The first three assumptions are valid to any degree of accuracy provided a small enough Δt is chosen. A Δt of about 15 minutes was used for the simulations. Smaller Δt's did not yield results that were significantly more accurate. As for the fourth assumption, a number of simulations were run to determine the most efficient direction to apply the spacecraft's thrust. It was found that thrusting in the direction of the spacecraft's velocity vector yielded the lowest ΔV and escape time. The last assumption is also reasonably accurate.

The algorithm for the Method is shown below. Initially, the Thrust, T, the radius of the low Earth circular orbit, R, and the mass of the spacecraft, M, are given. Also note that TA is the true anomaly angle, or angle from periapse, e and h are the spacecraft's energy and angular momentum per unit mass respectively, ΔA is the area swept out by the spacecraft's orbit during Δt, and β is the angle between the spacecraft's velocity vector and a vector tangent to a circular orbit at that location. The algorithm is:

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1) $V^2=\mu/R$, $\beta=0$	V for initial orbit
2) $h=RV$	h for initial orbit
3) $\Delta V=(T/M)\Delta t$	ΔV applied over Δt
4) $M=M-(T/g_0 I_{sp})\Delta t$	Consume fuel
5) $V=V+\Delta V$	Increase V
6) $e=V^2/2-\mu/R$	spacecraft energy
7) Is $e>4.351$ J/kg?	If so, end burn
7) $h=h+R\Delta V$	Spacecraft angular momentum
8) $a=\pm\mu R/(RV^2-2\mu)$	Orbit semi-major axis
9) $TA=\cos^{-1}[(h^2/\mu-R)/R(1+2eh^2/\mu^2)^{1/2}]$	Initial true anomaly
10) $\Delta A=(h/2)\Delta t$	Area swept out during Δt
11) $TA=TA+(2/R^2)\Delta A$.	Final true anomaly
12) $R=(h^2/\mu)/[1+\cos(TA)(1+2eh^2/\mu^2)^{1/2}]$	Final orbit radius
13) $V^2=\mu(2/R\pm 1/a)$	Final speed
14) $\beta=\cos^{-1}(h/RV)$	Final direction of travel
15) Return to 3)	

The + or - refers to hyperbolic or elliptic trajectories respectively. The orbit will be hyperbolic if $e>0$ and elliptic if $e<0$.

The algorithm begins with the spacecraft in low Earth orbit. The ΔV that the spacecraft will undergo during Δt is calculated and applied to the spacecraft impulsively. The new energy and angular momentum of the spacecraft are computed. The spacecraft's initial TA is calculated and, using Kepler's second law, the TA at the end of Δt is determined. From this, the spacecraft's velocity vector and orbit radius at the end of Δt are calculated. The routine runs until enough energy has been built up so

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that the spacecraft can escape the remainder of the Earth's gravity with an excess speed of 2.95 km/s.

This routine showed that the ΔV required was 9.504 km/s. The ratio of fueled mass to empty mass as therefore 1.274. The inerts for this system are 810 kg. Thus, the empty mass is 5,079 kg, and so the fueled mass is 6,472 kg. This is a far smaller craft than the chemically propelled one and since the requirements specify low mass, the SEP system was selected.

Launch Windows:

It should be noted that even if the spacecraft were in the Sun 100% of the time, it would still take about 320 days for the craft to perform the burn. In addition, the craft will spend some time in the Earth's shadow. During this time, the SEP system will be shut down. A quantitative method for modeling this added complexity was not possible with the computing power available, but this would add about another 30 days to the escape time while having a negligible effect on the ΔV required. This time requirement was considered an acceptable tradeoff because a time limit was not in the mission requirements and furthermore, with a little planning, it can be arranged for the airplane to arrive at Mars whenever it will be needed.

It takes 259 days for the spacecraft to reach Mars after Earth escape. The launch windows correspond to the launching dates such that Mars will be exactly on the other side of the sun from the Earth escape point 259 days after escape has occurred. Assuming a 350 day Earth escape time and an aerobraking time of 20 days (see Aerobraking and Re-entry), the launch windows in the first decade of the 21st century are as follows:

Earth launch	Earth escape	Mars capture	Low orbit arrival
June 4, 2002	May 20, 2003	Feb. 3, 2004	Feb. 23, 2004
July 23, 2004	July 8, 2005	Mar. 24, 2006	Apr. 13, 2006
Sep. 11, 2006	Aug. 27, 2007	May. 12, 2008	June 1, 2008
Oct. 30, 2008	Oct. 15, 2009	July 1, 2010	July 21, 2010

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The payload can be dropped anytime after achieving low obit, providing the weather and position of the orbiting payload permit.

Costing:

The cost of the mission was determined using the method developed by Science Applications International. This method provides algorithms for computing the number of man-hours to develop and produce a system based on the mass of that system. Some systems may already be developed, or may be modifications of a currently available system, and in these cases development costs were reduced accordingly. For the purposes of production costing, it was assumed that four vehicles will be built. The table below shows the inheritance classification and mass in kilograms of each system on the spacecraft (see Science Instrumentation for a description of abbreviations).

System	Block Buy	Exact Replica	Minor Mod.	Major Mod.	New System
Structure					228.1
Thermal control					99.2
Propulsion:					
SEP system		362.8			
Capture, deorbit, and circularization engines			120.1		
AACS	25.0				
Communications			42.0		
Antenna			19.2		
CDC					106.5
Aerobrake					405.0
Main solar arrays (2)	260.0				
Secondary arrays (2)	25.0				
Batteries	40.0				
Engine Power Units	187.0				
Science Instruments:					
VIMS		22.0			
GRS		25.0			
TES		11.0			
magnetometer		4.5			
PMIRR		25.7			
Radar Altimeter		24.5			
MOC		11.0			
AMI			26.3		

Total hardware mass=2,017.1 kg

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The required Development Labor Hours (DLH), Production Labor Hours (PLH), and Labor Cost (LC) for each system are shown in the table below:

System	DLH*	PLH*	LC**
Structure	220.9	318.8	8.10
Thermal control	130.8	244.4	5.63
Propulsion:			
SEP	130.9	810.2	14.12
Capture, deorbit, and circularization engines	309.8	299.2	9.14
AACS	0.0	193.2	2.90
Communications	229.5	552.5	11.73
Antenna	130.7	256.4	5.81
CDC	926.5	1,283.5	33.15
Aerobrake	544.7	375.3	13.80
Main solar arrays (2)	0.0	474.8	7.12
Secondary arrays (2)	0.0	114.9	1.72
Batteries	0.0	80.3	1.20
Engine power units	0.0	471.6	7.07
Science Instruments:			
VIMS	33.1	40.3	1.10
GRS	35.7	47.1	1.24
TES	21.8	23.6	0.68
Magnetometer	15.4	25.9	0.62
PMIRR	36.3	48.2	1.27
Radar Altimeter	48.9	255.5	4.57
MCC	10.7	63.3	1.11
AMI	196.1	278.9	7.13
Hardware Totals:	3,021.8	6,257.9	139.20
System support & GE	1,539.6	0.0	23.09
Launch+30 days O &GS	293.4	0.0	4.45
Science data dvlpmnt.	148.2	0.0	2.22
Program management	234.2	0.0	3.51
Flight operations	1,019.0	0.0	15.28
Data analysis	433.1	0.0	6.50
Mission Totals:	6,689.3	6,257.9	194.21

Total Mission Cost = 3.303 x LC = $641,470,000

*In thousands of hours **In millions of dollars

For estimating labor cost, it was assumed that the average rate of pay was $15 an hour. It was also assumed that the total cost of the project was 3.303 times the cost of the labor.

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Implementation:

The spacecraft design will be implemented according to the following time table:

Jan. 1989:
Begin planning of the spacecraft system. Make initial specifications as to how all subsystems will interact.

Jan. 1991:
Complete preliminary spacecraft system design.

March 1992:
Preliminary computer design completed.

May 1993:
Preliminary computer testing completed.

Nov. 1993:
Final computer design completed. Preliminary propulsion systems design completed.

July 1994:

Final computer design tested. Preliminary structure designed.

Jan. 1995:
Research on aerobrakes completed. Final propulsion systems design selected. Final computer unit assemble and tested. Preliminary communications system completed.

Jan. 1996:
Final Aerobrake, re-entry and communications systems, and structure designs selected. Preliminary power system design completed.

Jan. 1997:
Final power system design selected. Preliminary AACS design completed.

Jan. 1998:
Complete aerobrake and re-entry system, communications system, propulsion systems, and thermal control component testing. Assemble final structure for testing. Final AACS design selected. Final selection of science instruments. Redesign of the voyager antenna is completed.

July 1998:
Production of the radio transmitter and receiver is completed.

Jan. 1999:
Complete testing of AACS and power system components. Complete individual science instrument testing.

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Jan. 2000:

Integration and testing of the communications system, assembled structure and aerobrake, propulsion systems, power systems, attitude control system, thermal control systems, science instrument system, and the re-entry system is completed. Development of the flight software is completed.

Jan. 2001:

Assembly and integration of the spacecraft is complete.

Jan. 2002:

Spacecraft testing is completed.

There are some problem areas that have yet to be examined. The effects of the other planets in the solar system on the spacecraft have been neglected in this analysis as has the the effect of the Moon on the escape trajectory. The effects of the dark periods must be studied so that a lauch date can be planned such that the spacecraft will escape the Earth in the proper direction. Finally, the effects of the spacecraft drifting out of Earth solar orbit during the SEP burn must also be studied.

Summary:

This completes the mission planning process. A summary of the results are as follows:

1) The ΔV to escape the Earth is 9.504 km/s.

2) The spacecraft's hyperbolic approach velocity to Mars is 2.67 km/s.

3) The ΔV to deorbit the payload is 116 m/s.

4) The cost of four missions is estimated to be $641.47 million.

In addition, launch windows have been computed and an implementation plan has been presented.

For Earth escape, Chemical and SEP propulsion units were compared and the SEP system was chosen because it resulted in a lower spacecraft mass.

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Final Note:

It should be pointed out that the RFP requirements specify a system that fails to utilize the full advantages of the SEP system. If the aerobraking requirement were dropped, the SEP system could be used to spiral the payload into Mars. The total ΔV for Earth escape and Mars capture would be 14.95 km/s. The spacecraft, less tanks, would weigh 3,406 kg. Using a tankage factor of .18, this spacecraft would be 5,442 kg instead of 6,472 kg, a 16% reduction in mass. The mission duration would remain unchanged, for although it would take 100 days to capture into low orbit, the spacecraft would escape Earth 80 days sooner. Other advantages include completely non-explosive fuel, more reliable capture engines, a reduced dependency on actuators, and no dependency on aerobraking, a technique that is, thus far, untried. In addition, the entire system arrives at Mars where it could be refueled and relaunched on other missions.

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STRUCTURES

The project objective is to develop a conceptual design for the spacecraft system required to deliver the components of a manned aircraft to the Martian Surface. The requirements are to provide support of components, layout of componets,and thermal control of components. The spacecraft is able to be retrived by a remote manipulation device,and fit in the payload bay of the Space Shuttle. In particular, the structure subsection is responsible for component layout, justification,mass/inertia configuratio, deployment issues, shuttle launch compatibility/on orbit assenbly, materials, thermal control considirations, and subsystem interactions with all the other subsections.

Some of the charactiristics required of new materials for spacecraft applications are resistance to stress corrosion cracking (SCC). In orbit, the materials will need to survive the space enviroment, specifically, atomic oxygen which can damage thermal-optical properties,solar ultraviolet radiation,and microcraking in composites. Materials need to posses low-outgassing-under-vacuum properties. Some of the materials that we looked into are not suited for space. For instance, polymers are very susceptible to ultra violet radiation, due to their high absorption coefficient.

Composites such as graphite-epoxy have been choosen over titanium in some applications because of their superior mechanical(stiffness, strength, fatigue,and buckling stability) and thermal properties. Stiffness is an important property for spacecraft structures to withstand high shear stresses and bucking stresses in compression. Several graphite-epoxy materials were considered; high tensile strength,high modulus, and ultrahigh modulus. We also considered two types of staking sequences. Type I[0,+/-30], type II[0,90,+45,-45,90,0]. In addition, some material properties of Ti-6Al-4V are provided below for comparison. The type II laminate and ultrahigh modulus yielded values of both longitudinal and transverse stiffnesses that were greater than the values for titanium. In addition, we can obtain a weight savings of 50 percent when graphite epoxy is used in place of titanium for structural applications.

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Furthermore, because type II laminates and ultrahigh modulus materials had values in stiffness that exceeded the values for the titanium alloy, the P 75 (2034D) ultrahigh modulus material was selected.

Material type	Laminate Type	laminate Density(lb/in^3)	Moduli(x10^6 psi) Elasticity		Strengths (x10^3 psi) Tensile		Compressive	
			x	y	x	y	x	y
HTS	I	.057	12.4	2	91	9	65	16
(T300/5208)	II	.057	8.7	8.7	89	8	123	123
HMS	I	.058	15.3	1.7	74	8	56	14
	II	.058	12.3	12.3	80	80	123	123
UHMS	I	.061	21.3	1.9	68	8	60	10
(2034-D)	II	.061	18.7	18.7	78	78	112	112
(P-75)**								
Ti-6Al-4V		.16	16.5	16.5	130	130	130	130

** *selected*

In additon, metal alloys were also considered for the material makeup of the spacecraft. Specifically, beryllium, aluminum-lithium, Al-Fe-Ce, and Al-Fe-Si were considered. Beryllium has a number of applications where the high stiffness of the material can be used to optimise weight and volume, and also raise the natural frequency of the structure to avoid resonance problems. In addition, beryllium can be used for antenna pointing mechanisms as well as solar array drive mechanisms. Because of the high melting point (1278 C) and low density (1.85 g/cm^3) berylium can also be used for the nose cap of the aerobrake structure. Berylliym has a modulus to density ratio 6.4 times greater than aluminum. In addition, beryllium 's high strength to density ratio, its high specific heat, and high thermal conductivity proved to be the best choice in comparison to Al-Fe-Ce and Al-Fe-Si.

Aluminum-lithium alloys have a 5-20% increased modulus of elastisity and an 8 to15% reduced density compared to present spacecraft aluminum strutural alloys . Furthermore, the cost of Al-Li may soon be cut below that of conventional aluminum because the alloy can now be formed superplastically. In addition, aluminum alloys such as Al-Fe-Ce and Al-Fe-V-Si have demostrated superior elevated temperature strength over conventional

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aluminum alloys. Today, Al-Li alloys are candidate materials for weight-critical structural applications because of their high stiffness to weight ratio. The additon of every 1 weight percent of Li causes the stiffness to increase by 6 percent while reducing the density by 3 percent. Unfortunately, these alloys exhibit poor fracture toughness levels when compared to convetional Al alloys.

Because of the different conditions the materials will go through in space, materials are required to have a high degree of dimensional stability, low weight, high strength and stiffness. In low earth orbit, materials will be subjected to repeated thermal cycles from about 200^{o}F to -210^{o}F, to ultraviolet radiation,and to ultra high vacuum. In higher orbits such as Geosynchronous Earth Orbit, materials will be subjected to large doses of high energy electrons.

Protective coatings on the spacecraft are needed to maintain it within the design temperature limits and increase its life time and reliability. High temperature materials are needed to protect the spacecraft from the high temperatures that are encountered during reentry. The protective coatings that we considered for atomic oxygen sensitive materials were kapton, indiom tin oxide, gold, platium, Aluminum dioxide, silicon dioxide, and polytetrafluoroethylene (PTFE). Monolithic ceramics have good temperature strength and good oxidation resistance, but they are brittle and currently have low reliability,which is why we did not select them for our spacecraft. Ceramic matrix composites have the benefits of good high temperature strength, toughness, and oxidation resistance , and a maximum operating temperature of 3000^{o} F. The major problem with ceramic matrix composites is the fabrication of a dense body with uniform distributred fibers that give the desired material properties . This is why ceramic matrix composites were not chooen for our spacecraft.

Basic requirement for thermal control coatings is to keep the spacecraft components within the allowed temperature limits . In addition, low emittance coatings are required for the composite structure to reduce the extent of cooldown during solar occult, protective coatings used in GEO must be able to withstand high energy electrons and protons in addition to ultra- violet radiation, and a higher electrical conductivity is needed in GEO to eliminate spacecraft charging.

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COMPOSITE STRUCTURE GEO		MANNED HABITAT LEO
OPTICAL PROPERTIES	SELECTABLE WITH Æ < 0.3	SELECTABLE WITH Æ > 0.8
TEMPERATURE	-100 TO +80 C	-100 TO +40 C
ENVIROMENT	UV, e-,p+,VAC,dT	UV, VAC, dT
ELECT. CONDUCTIVITY	<10^-8(ohm^-1 cm^-1)	10^-8 to10^-17(ohm^-1-cm^-1)

Æ IS THE ABSORPTANCE

There are however some limitations in thermal control coatings , these are as follows.

RADIATOR COATINGS (LOW ABSORPTANCE, HIGH EMITTANCE)

COATING	PERFORMANCE
S -13 GLO(ZnO/RTV-602)	RTV 602 DISCONTINUED, NEW SILICONE BINDER FOR QUALIFICATION
ZOT (Zn_2TiO_4/SILICATE)	ABSORBS MOISTURE, HARD TO CLEAN
Z-93 (ZnO/SILICATE)	ABSORBS MOISTURE, EASILY CONTAMINATED
Al OR Ag/TEFLON	LARGE AREA APPLICATION DIFFICULT
WHITE SILICATE PAINTS	DIFFICULT TO APPLY AND CLEAN
SILICON CARBIDE	VERY EFFECTIVE FOR RCC, ACC
TEOS	INCREASES OXIDATION PROTECTION OF SILICON CARBIDE

STRUCTURAL COATINGS (LOW ABSORPTANCE, LOW EMITTANCE)
NO QUALIFIED PAINT-TYPE COATINGS AVAILABLE

Carbon-carbon composite materials offer the benefits of good strength and toughness at very high temperatures (up to 2500^{o}C). They also have a very low density, less than 25 percent of the density of superalloys and 50 percent the density of monolithic ceramics and ceramic matrix composites . Carbon-carbon components have a maximum operation temperature of 2500^o C which is much higher than that of ceramic matrix composites and monolithic ceramics. Carbon-carbon composites do have some draw backs, such as oxidation contamination and poor quality in fabrication. However, the oxidation can be terminated by applying protective coatings. Carbon-carbon composites have high strength potential, ultimate stress > 40,000 psi, E >10^7 psi,high thermal and chemical stability, T_{MP} > 7400 °F, K ~ 60 BTU -in/hr - ft^2 - F, absorptance ~ 2×10^{-6} $^oF^{-1}$,low density, density < 0.07lbm/in^3 .

Because of the research in the development and improvement of aerospace

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materials, new materials have been introduced for aerospace applications. One of these materials is reinforced carbon-carbon (RCC). Reinforced carbon-carbon may be used for thermal protection systems,hot structures, and engines. For thermal protection systems we looked at several possibilities such as the shuttle Orbiter thermal protection system composed mostly of silica-based ceramic tiles. We also looked at oxide ceramics, coated refractory metals, bulk graphite, graphite-epoxy, reinforced carbon-carbon (RCC), advance carbon-carbon (ACC). During reentry the spacecraft will be exposed to temperatures up to 3000 °F, this is beyond the capabilities of the silica tiles. Thus they will not be used for our spacecraft. RCC, ACC, and graphite-epoxy were the prime candidates because of their superior mechanical and thermal properties. These are as follows, maintance of reproduceable strength levels to 3000 °F,sufficient stiffness to resist flight loads and large thermal gradients, low coefficient of thermal expansion to minimize induced thermal stresses, oxidation resistance sufficient to limit strength reduction, and tolerance to impact damage. In orbit, graphite-epoxy serves primarily to remove heat generated by science instruments and withstand thermal loads imposed by differential solar heat.

Carbon-carbon materials have high thermal conductivity and low thermal expansion . Because of these properties, carbon-carbon materials are very resistant to thermal shock. They also have a low density , about 70% of that of aluminum alloys, which is about .065 lbm/in^3 .

Since the strength properties of carbon-carbon does not degrade with increasing temperature. The properties given below for room temperature also apply for temperatures as high as 4000 °F.

MATERIAL	RCC	ACC-3	ACC-4 **
FIBER	WCA(RAYNON)	T300(PAN)	T300(PAN)
WEAVE	SQUARE	8-HARNESS	8-HARNESS
ORIENTATION	X-PLY	QUASI-ISOTROPIC	0°(WARP)
IN PLANE			
TENSILE STRENGTH ,KSI	7.5	16	44
TENSILE MODULUS,MSI	2	11	16

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COMPRESSIVE STRENGTH,KSI	24	16	28
COMPRESSIVE MODULUS	4.4	11	16
SHEAR STRENGTH, KSI	1.8	1.0	2.0
OUT OF PLANE			
TENSILE STRENGTH, KSI	0.8	0.4	0.5

** SELECTED

In oxidizing enviroments, carbon-carbon parts must be coated and sealed with silicon carbine. Because of the differences in thermal expanxion between the silicon carbide and the carbon-carbon part, the coating develops microcracks. However, these cracks can be impregnated with with tretaethylorthosilicate (TEOS). This process leaves silica (SiO_2) in all of the microcracks, enhancing the oxidation protection of the carbon-carbon substrate.

Applications for advance carbon-carbon include thermal protection systems and spacecraft structure. Because the primary structure which can be made of ACC which can operate at very high temperatures,no insolation would be required. If ACC can be developed for these retrofitable control surfaces this would result in a significant weight reduction over the conventional two skin construction. For instance, a weight savings of 5000 lbms would be obtained for the Space Shuttle. This is why ACC has been selected to be one of the principal materials for our spacecraft.

Studies done on the space shuttle have shown that new oxidation resistant coatings can decrease the mass loss during reentry heating cycles. It was found that for a simulation of 100 missions, a mass loss of .02 lbm/ft^2 for the RCC was obtained,whereas oxidation resistance of the coated and sealed ACC was twice as good as the baseline RCC. Furthermore, the addition of a low-temperature glass former significantly reduces mass loss. The mass loss for three carbon-carbon materials, RCC/SIC/TEOS, ACC/SIC/TEOS/MAP and, ACC/DSIC/TEOS/MAP in a study showed that the addition of a low temperature glass former significantly reduces the mass loss. Furthermore, the oxidation resistance has been improved by modifying the baseline silicon carbide coating (DSIC). This dopped and sealed coating has 25 times the oxidation resistance of the Shuttle baseline RCC material.

The bounding of thin carbon-epoxy face skins onto aluminum honeycomb core provides for an

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electrically conductive highly stable surface profile, even under thermal cycling under vacuum. This another reason why graphite epoxy composites were choocen. Other possibilities that were considered were antennas with kevlar skins with dipole crosses and a honeycomb core.

Because we used hydrozine as a propellant, we employed tanks that are made of ti-6Al-4V since it offers a greater weight savings than 304 L stainless steel. Furthermore, there are only a few materials (titanium and stainless steel) that are fully compatible with the hydrozine propellant. Other materials that were considered for structural use were fiber-reinforced plastic composites. They reduce weight and increase the stiffness of structural components. In particular, Kevlar-49 Reinforced Plastic (KRP) and polyurethane shock absorbing foam in a sandwich configuration have proven to be effective in protecting the walls of spacecraft from impact of hypervelocity meteroids traveling at speeds up to 60 km/s, thus this combination of protective materials will be used for the spacecraft. Recent advances and expected future improvements in advance carbon-carbon include,matrix improved strength and stiffness and improved hybrid construction. Improvement in oxidation resistance coatings and improved sealants. Future applications include thermal protection systems, and hot structure applications. Current research in silicon carbide coatings and advance coating sealants is expected to yield additional improvements in matrix oxidition protection and increase maximum use temperature.

Finally, we selected graphite-epoxy for the instrument bus as described above, ACC for the primary structure because of the properties given above. RCC was seclected for the aerobrake structure with a nose cap of beryllium. Ti-6Al-4-V was choocen for the fuel tanks becuase we using hydrozine as fuel. We selected protective coatings such as silicon-dioxide for oxidation protection for ACC. This coating greatly reduces the mass loss during reentry, for redundancy we added to the exterior walls kevlar -49 Reinforced Plastic (KRP) and polyorethane shock absorbing foam to protect the spacecraft from impact of meteroids.

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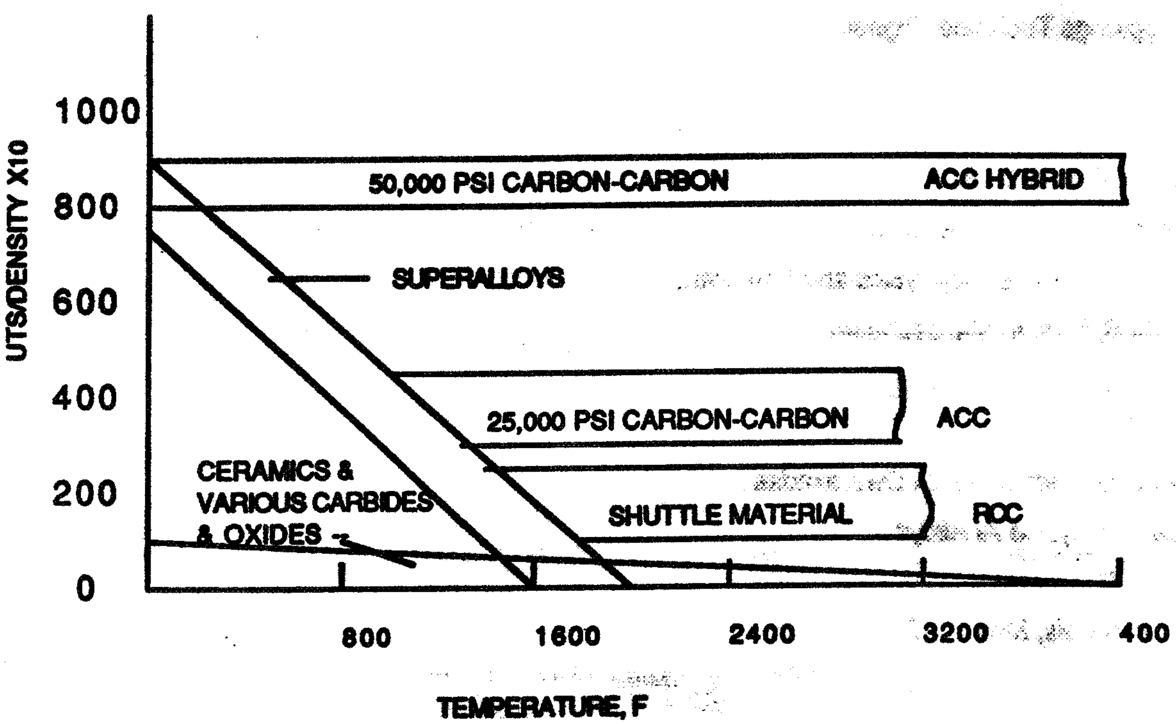
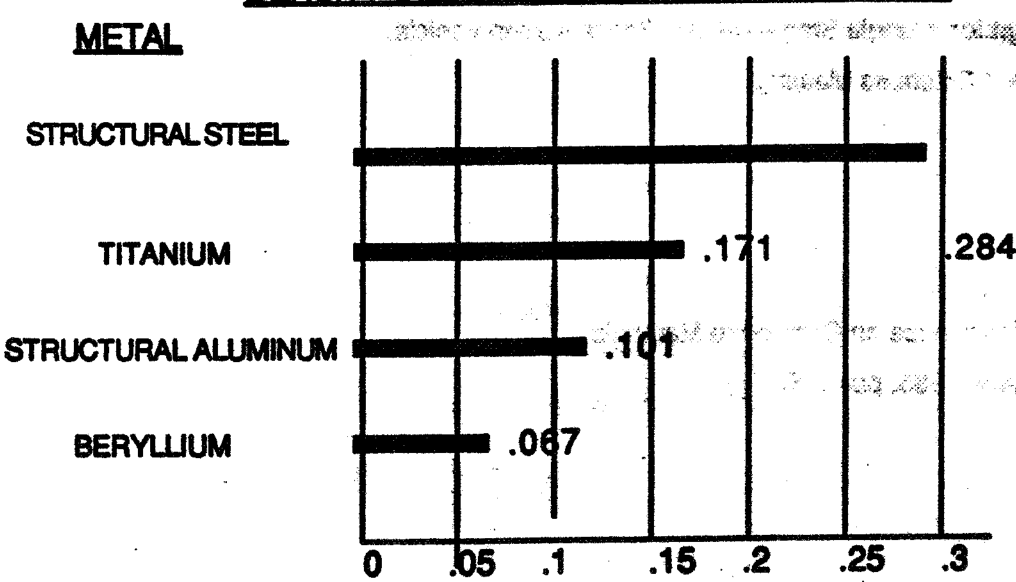
STRENGTH EFFICIENCY OF MATERIALS



ACC is twice as strong as RCC , ACC is made up of woven carbon cloth . If unidirectional carbon fiber tapes are interplied with woven cloth to create hybrid ACC,this can increase the strength in at least one direction by 50,000 psi

DENSITY CONPARISON OF VARIOUS METALS



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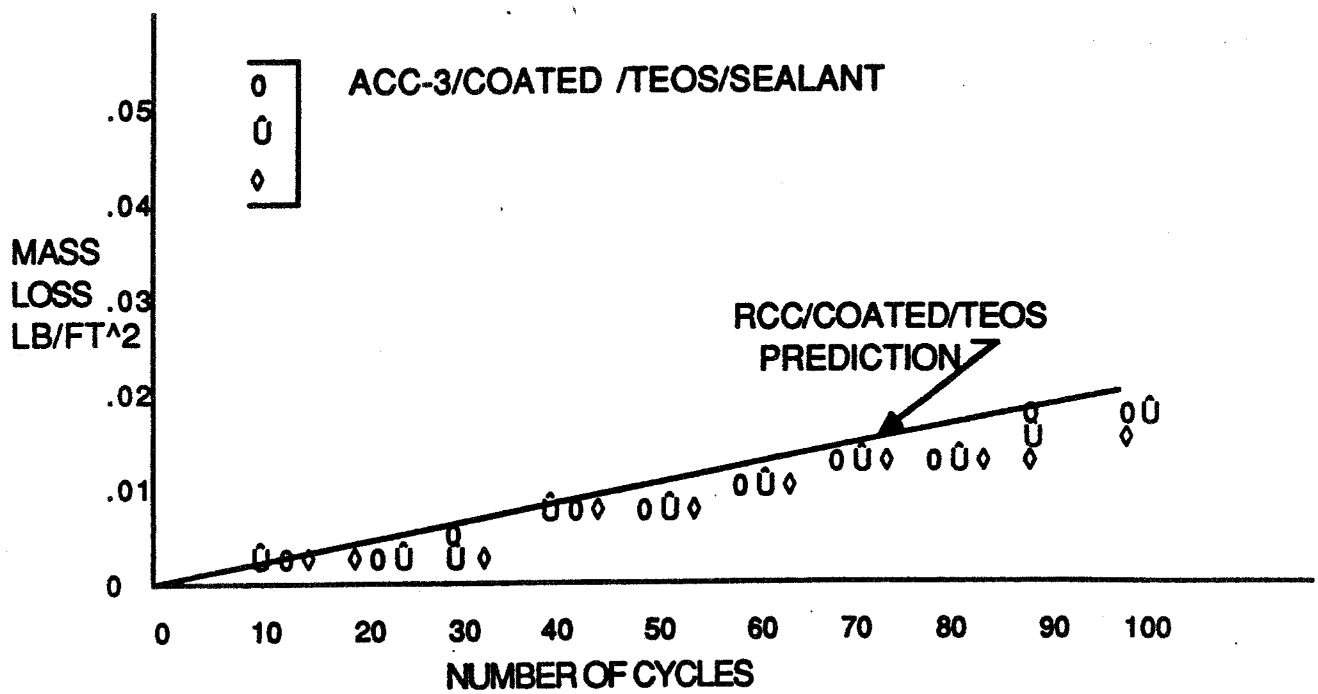
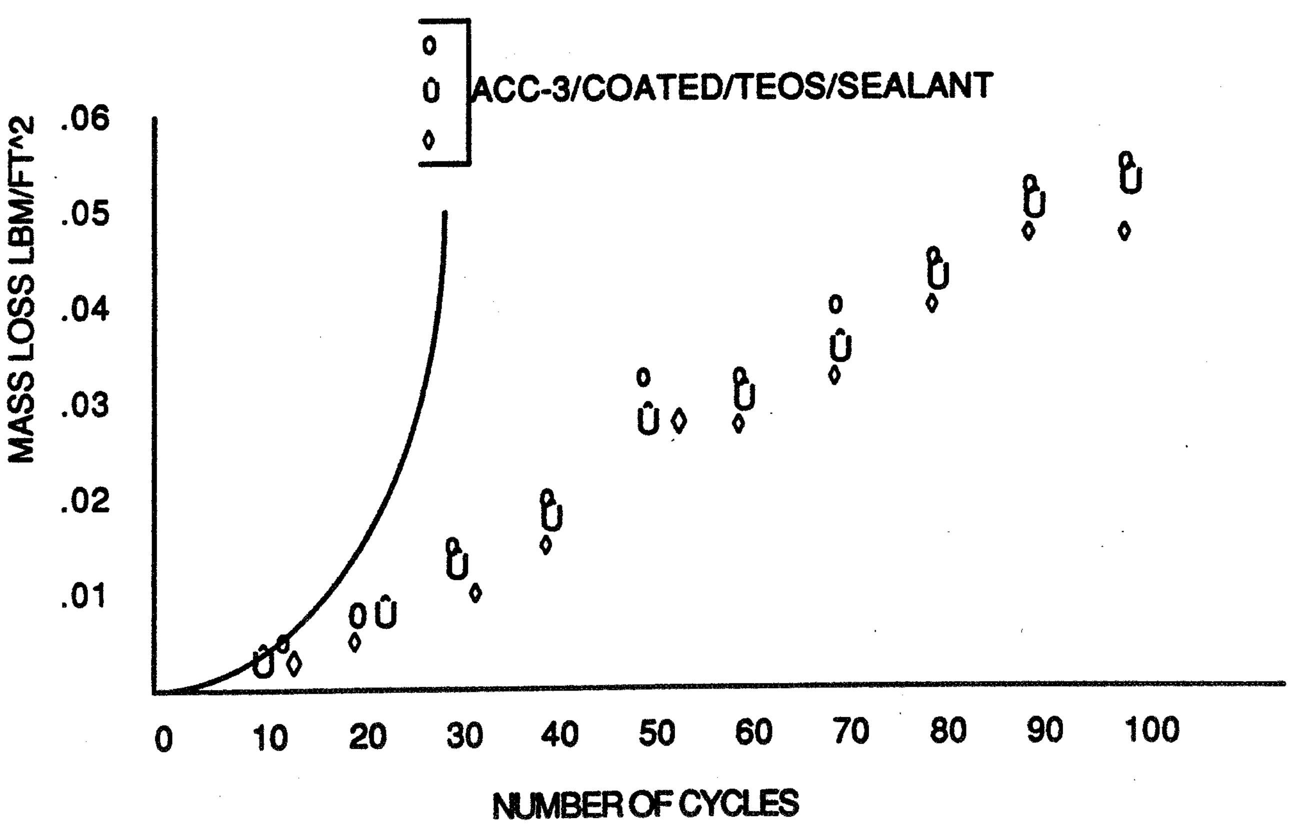
IMPROVED OXIDATION RESISTANT COATINGS



OXIDATION RESISTANT COATINGS



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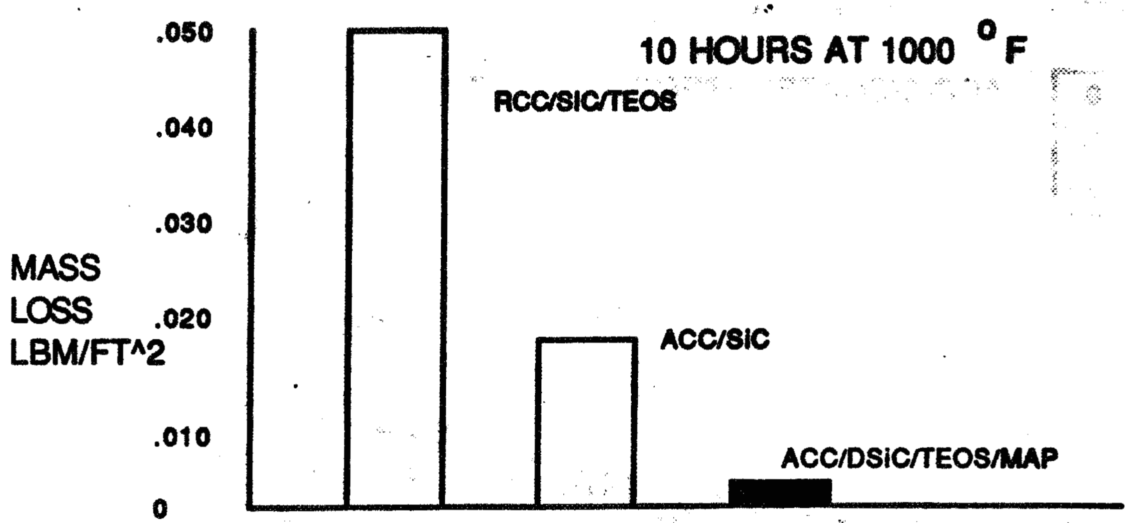
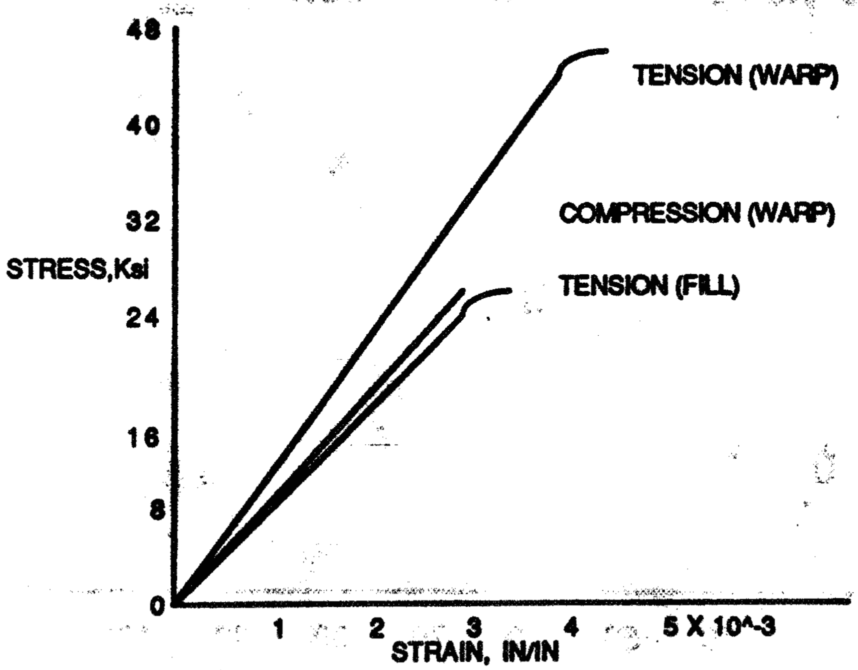
EFFECT OF COATING/SEALANT ON MASS LOSS



USE OF SILICON CARBIDE COATING FOR OXIDATION PROTECTION OF ACC

THIS DOPED AND SEALED COATING HAS 25 TIMES THE OXIDATION RESISTANCE OF THE SHUTTLE BASELINE RCC MATERIAL

2-D ACC-4, 20 PLY, (0°,WARP)



TYPICAL 2-D CARBON-CARBON STRESS-STRAIN BEHAIVOR

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MATERIALS	BENEFITS	PROBLEMS	SELEC TION
GRAPHITE/ EPOXY	HIGH PERFORMANCE LIGHT WEIGHT HIGH DEGREE OF THERMAL STABILITY ASSURED ACCURACY FOR SENSITIVE INSTRUMENTS SUPERIOR MECHNICAL (STIF FNESS, STRENGTH,FATIGUE) THERMAL PROPERTIES	OFFFER POTENTIAL OF ZERO OR NEG COEFFICIENTS OF THERMAL EXPENSION	YES
Ti-6Al-4V	USED INSTEAD OF HIGH STR ENGTH AL BECAUSE OF HIGH HIGH TEMP STRENGTH,FRAC TURE TOUGHNESS,FOR PROP ELLANT TANKS, COMPATIBLE W/ HYDROZINE		YES
Al-Li	HIGH STIFFNESS TO WT RATIO Li INCREASES STIFFNESS BY 6% REDUCES DENSITY BY 3% 5-20% INCREASE IN MODULUS OF ELASTICITY ,COST WILL DECREASE COMPARED TO CONVENTIONAL ALUMINUM	FAILS HIGH TEMPERATURES	NO
KEVLAR 49 REINFORCED PLASTIC AND POLY-URETHANE	SHOCK ABSORBING FOAM PROTEDTION OF METOIDS		
MAGNESIUM		LOW RESISTANCE TO BOTH SURFACE CORROSION AND SCC	NO
BERYLLIUM	HIGH SPECIFIC STIFFNESS RAISES NAT FREQUENCY OF STRUCTURE TO AVOID RESONANCE PROBLEMS HIGH MELTING POINT (1287 C) LOW DENSITY (1.85 g/cm)	TOXIC	YES
METAL OXIDE COATINGS		DO NOT PROVIDE COMPLETE PROTECTION AGAINST LONG TERM ATOMIC OXYGEN DIFFUSION	NO

Aerobrake

The conceptual design is a manned Mars aircraft space delivery system. The requirement for the Aerobrake subsystem is to design the reentry vehicle and the Aerobrake heat shield. The shield is used to protect the spacecraft during aerobraking maneuvers and the payload during reentry. The design should consider the use of off-the- shelf technology and not use technology available after 1998. Also, the vehicle should have a minimum lifetime of four years. Furthermore, the design should minimize mass and cost and to be reliable and simple.

Aerobrake Requirements:

1. Dissipate orbit energy in atmosphere.
2. Protect the aircraft from launch to surface of Mars.
3. Send telemetry to communication, command, and control.
4. Accept commands from communication, command, and control.
5. Relay satellite separation control
6. Control outputs from aerobrake attitude, separation, and parachute control
7. Sensor inputs for temperatures
8. Components for structure, thermal protection system, parachute and landing.

Aerobrake Shape

Aerobrake shapes considered is between raked elliptic-cone and raked sphere-cone. The study criteria took account of navigability, stability and control, afterbody heating, radiative heating convective heating, frustrum edge heating, multiple pass. The raked sphere-cone was chosen over the raked elliptic-cone. Both conic shapes rated

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well in all of these criteria except for raked elliptic-cone shape which has a problem with frustrum edge heating.

Aerobrake shield and payload trade studies

Aerobrake size study considered a shield with two payload cargobays each18m long and 4.5m in diameter. The reason for two cargo bays is due to the cargo's (Mars Airplane)large dimension. The first study considered placing two cargobays end to end against the shield (figure 1) but was ruled out due to instability of the configuration during aerobrake and reentry. If a shield is made larger to accomodate the length then the result is a larger and heavier shield of well over 400 m^2 and 2000 kg. A larger shield is extremely elastically unstable.

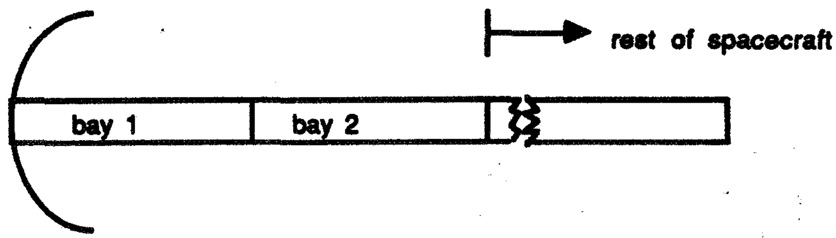

figure 1 Payloads stacked end to end

The second configuration is two bays of length 18 m and 4.5 m in diameter which are laid down on their sides against the shield (figure 2). An advantage is a more stable configuration for aerobraking and reentry incomparison to first design. This requires a larger shield area in the range form 243 m^2 to 425 m^2. An area of that size will result in a massive shield of 2000 kilograms but can protect the spacecraft from aerodynamic heating. However, a shield size of 243 to 425 m^2 will under go elastic instabilities during aerobraking and reenty.

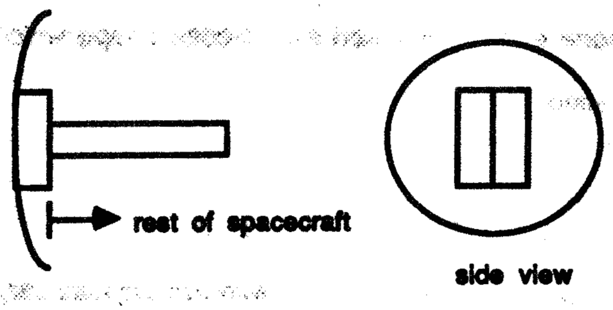

figure 2 cargo bays laid side to side against shield

The chosen configuration (figure 3) is two cargobays 18 m long and 4.5 m in diameter divided in half and stack on top of each other and laid down against the shield. This will result in a smaller shield area of 154 m^2 and a decrease of mass to 405 kilograms. Also, the decrease in shield area will increase the elastic stability of the shield during aerobraking and reentry.

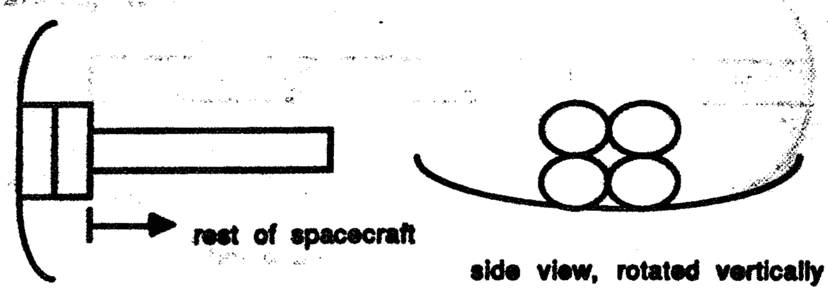

figure 3 four cargo bays, two bays stacked on top of two bays

Separation

After orbit circulization, the reentry vehicle will detach from the science bus by explosive bolts. Then thrusters will separate the reentry vehicle from the bus in order to fire deorbital engines. The acceleration chosen as 0.5 m/s^2 using minimum fuel of 5.89 kg at a Δv of 7.5 m/s^2 for a 35 second maneuver. The Δv was calculated using the

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Δv equation $\Delta v = g_o I_{sp} \ln(M_i/M_f)$ where the M_i (initial mass) is 2312.4 kg, I_{sp} is 300 seconds, and g_o is 9.8 m/s^2. Profile separation maneuver is shown belowin figure 4.

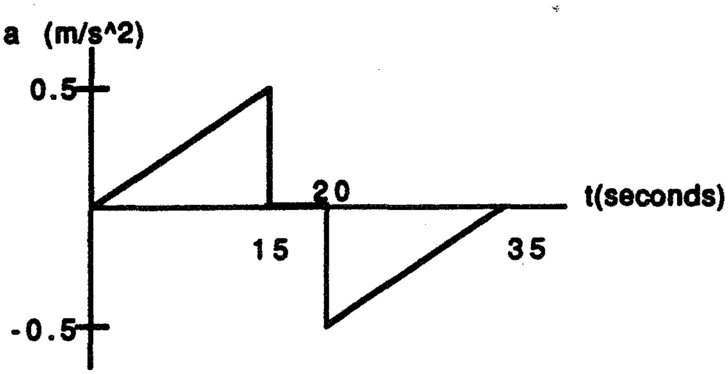

figure 4. acceleration profile of separation

Deorbit transfer and Deorbital engine

Deorbit transfer will begin from 500 km above surface to 244 km at atmospheric entry. Trajectory for deorbital Hohmann transfer has been calculated by Mission Planning subsystem. The resulting Δv is 116 m/s and the I_{sp} for deorbit engines is 300 seconds with an initial reentry mass of 2312.4 kg. Using Δv equation and solving for final mass (M_f) which is 2223.03 kg therefore, amount of fuel needed for this maneuver is found to be 89.37 kg. The tank size is 52.2 cm in diameter containing Hydrazine of density 1200 kg/m^3. The total time for the deorbit transfer is found to be 6674 seconds.

Reentry Method of Attack and Results

Reentry is performed by numerical methods to simulate reentry into Martian atmosphere. The initial parameters for the code are μ (Gm), R (radius of planet), ∂_o (atmospheric surface density), ∂ (density), ø (angle of reentry vehicle), ß (altitude-density constant), A (spacecraft frontal surface area), A_p (parachute area), m (mass of spacecraft), C_D (coefficient of drag), D (drag on shield), Dp (drag from parachute), Δt

(time increment), v_{xi} and v_{yi} (initial x and y velocity components, respectively), x_i and y_i (initial x and y position components, respectively).

The algorithmic code is as follows:

$ø = \arctan(v_x/v_y)$	angle of reentry vehicle
$v = \sqrt{v_x^2 + v_y^2}$	magnitude of velocity
$r = y + R$	distance from center of planet
$\partial = \partial_o e^{-\beta y}$	density function
$D = 1/2\, C_D \partial v^2 A/m$	drag acceleration of spacecraft
$D_p = 1/2\, C_{Dp} \partial v^2 A_p/m$	drag acceleration of parachute
$W = -\mu/(r)^2$	weight acceleration
$C = v_x^2/r$	centripital acceleration
$a_x = -D\cos ø - D_p \cos ø$	x-component of acceleration
$a_y = C + W - D\sin ø - D_p \sin ø$	y-component of acceleration
$x = x + v_x \Delta t + 1/2\, a_x \Delta t^2$	x-component of position
$y = y + v_y \Delta t + 1/2\, a_y \Delta t^2$	y-component of position
$v_x = v_x + a_x \Delta t$	x-component of velocity
$v = v_y + a_y \Delta t$	y-component of velocity
$t = t + \Delta t$	time increment

The code simulates a reentry vehicle entering the atmosphere at 244 km above the surface and traveling at a velocity of 3440 m/s at an angle of -3.83999 °. The program is given a predetermined optimal height of 6000 m to deploy a chute of 24 m in diameter. At this altitude the vehicle is traveling approximately 70.5 m/s or Mach 0.30 at angle of -90 ° with an acceleration of about 4.43 m/s^2. The program keeps iterating the code down to the surface giving the time, angle, velocities, and acceleration of the vehicle at impact up to the surface.

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An altitude of 6000 m was chosen based on the velocity the reentry vehicle in comparison to the Viking at the same altitude. At 6000 m the Viking velocity is 4800 m/s while our reentry vehicle is traveling a mere 70.5 m/s because the frontal area of our spacecraft is 154 m^2 while the Viking area is about 2.22 m^2. Therefore, Viking area produces extremely less drag in contrast to the reentry vehicle which has a greater deceleration than Viking.

Parachute diameter is found by using the equation $v_t = \sqrt{2mg/C_D \partial A}$. Since the altitude is 6000 m then the ∂ (density) can be determined by using the density function found in algorithm code. The known variables are m, g, A, and C_D for the reentry vehicle. Therefore, the values for the vehicle are m=2114.7 kg, g=3.724 m/s^2, A=154 m^2, and C_D=1.75 then v_t= 106 m/s. Therefore, we know the velocity at which the vehicle will be terminal is 106 m/s and the parachute coefficient of drag is C_{Dp}=.58 and substituting into the same equation and then solving for A (area). Therefore, the result is a 24 m diameter parachute. This value is used in the algorithm.

From an altitude of 6000 m at 70.5 m/s the vehicle decelerates to 37 m/s or Mach 0.154 at an altitude of 1500 m then the chute is separated and the part of bottom shield is separated, exposing the landing gears and retro-rockets. Since the simulator does not take into account the retro-rockets of the vehicle then back of the envelope calculations are used to solve for acceleration and time from 1500 m to the surface. Assuming when retros are fired, descent is a constant acceleration and parameters at parachute separation are initial conditions at 1500 m and final conditions are at the surface. Also, assume that velocities and accelerations are zero in the x-direction since the angle of vehicle is 90 degrees. Then constant acceleration and time can be solved using free fall equations: $y=y_o+1/2(v_{yo}+v_y)t$ and $v_y=v_{yo}+a_yt$, where y_o and v_{yo} are initial conditions and y and v_y are final conditions. Initial values are velocity is -37 m/s and altitude is 1500 m and final values are velocity is zero and altitude is zero.

Then solving, time is 81 seconds and acceleration is 0.456 m/s^2 from an altitude of 1500 m. The entire landing sequence is monitored by a radar altimeter which inputs altitude readings and outputs readings to parachute and landing gear and retros. Then a free body diagram is made and a thrust equation is derived.

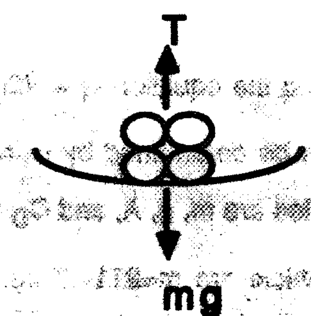

figure 5. free body of reentry vehicle with retros.

Using Newton's equation (F=ma) we derive a thrust equation T=m(a+g). Since the value of acceleration is known then the initial thrust of the retros is calculated to be 8840 N. Then using Δv equation assuming specific impulse is 300 seconds a final mass and fuel mass is computed as 2088.3 kg and 26.45 kg, respectively. Also, assuming fuel is hydrazine then density is 1200 kg/m^3 therefore, fuel mass divided by fuel density equals volume of fuel .022 m^3. Then from volume of sphere then radius of tank can be solved to be 0.174 m, therefore tank diameter is 34.79 cm.

<u>Power requirement and supply</u>

Reentry will be on self power for 7919 (T_E) seconds after separation. The number of cells need for a battery can be calculated by dividing stored energy by watt-hours/cell. The battery used for the vehicle is a nickel-cadmium with stored energy 2444 W/hr by assuming that the bus voltage (V) is 35 V and power load (P_L) is 500 W.

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The energy per battery cell is 32 W-hr/cell. Therefore, the number of battery cells required is 76. The battery capacity ($C=P_L T_E/DODxV$) Where the Depth of Discharge is 45% therefore, battery capacity is 341 ampere/hour. Battery weight is the stored energy divided by watt-hour/kg and for nickel-cadmium watt-hours/kg is approximately 25 therefore, battery weight is 97.8 kg.

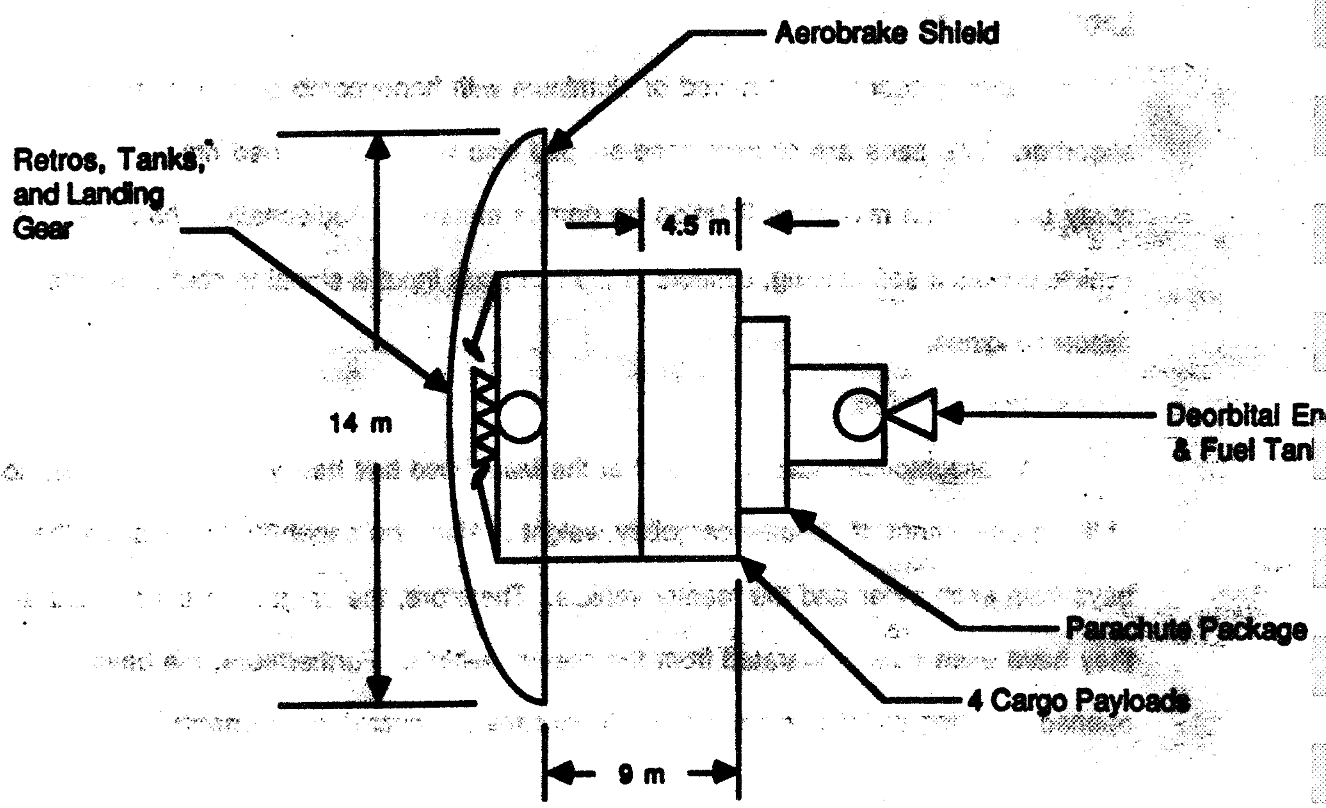
Landing gear

Landing gear is constructed of aluminum with honeycomb aluminum shock absorber. The pads are circular cone-shaped disc with down-turned rim to prevent soil spray and to gain maximum flotation on porous material. Additionally, when the reentry vehicle makes a soft landing, sensors in the foot pads input a signal to shut down the descent engines.

Cargo bay opening

An assumption must be made that the Mars base has heavy lift machinary capable of lifting equipment of Shuttle cargobay weight . Also, the capability to separate the bays from each other and the reentry vehicle. Therefore, the cargobays are opened after they have each been separated from the reentry vehicle. Furthermore, the bays are opened by lifting the upper section which exposes the aircraft components.

Aerobrake Component Layout



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Attitude and Articulation Control Subsystem (AACS)

Introduction:

Before a design for the attitude control system can be made, an understanding of what it does is necessary. The attitude control system is responsible for pointing the spacecraft in the proper direction and keeping the spacecraft on course, by using sensors, a computer, and a means of correcting any course errors. It is also responsible for small delta V maneuvers.

In order to simplify the design for the AACS, we broke the mission into four distinct phases. First an earth spiral out, this is where the ship builds up enough speed to break free of the Earth's gravity. Second an Earth to Mars transfer. Third the aerobrake maneuver, where the ship dives into the Martian atmosphere to lose speed. Finally the Mars orbit, where the instrument bus orbits Mars, for several years.

The next step is to determine the forces that act on the space craft. These forces are: solar radiation, gravity-gradients, aerodynamics, magnetics, meteoroids, debris, internal moving parts, thrust misalignment, thermal emissivity, electromagnetic radiation, and propellant leakage. Some of these are negligible, when compared to the others. Also the major forces will be different for each part of the trip.

Certain forces are difficult, or impossible, to determine in a preliminary design, like this, they include; magnetics, meteoroids, thrust misalignment, thermal emissivity, EM radiation, and propellant leakage. These effects have been ignored, in our design, but are probably minor compared to the others, so this should not be a problem.

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Analysis:

The analysis of the AACS system will be carried out in four parts, one for each phase. Only the most important effects will be considered in each phase.

The forces acting on a space craft produce torques. These torques in turn produce an angular acceleration, and a rotation. The job of the attitude control system is to stop this motion, reverse the direction of motion, then stop the motion in the correct place.

To stop the motion,

$$((\text{alfa})\ t)_{\text{disturbance}} = ((\text{alfa})\ t_1)_{\text{control}}$$

Where alfa is the angular acceleration, and t is the time involved. The time to stop motion is then given by,

$$t_1 = ((\text{alfa})\ t)_{\text{disturbance}} / ((\text{alfa})\ t)_{\text{control}}$$

The amount of turning of the spacecraft is given by;

$$\text{Theta} = (0.5(\text{alfa})\ t^2)_{\text{disturbance}} - (0.5(\text{alfa})\ t_1^2)_{\text{control}}$$

The attitude control system must then repoint the spacecraft. In order for the final rotation to stop as the correct angle is obtained, half the time must be spent accelerating, and half decelerating. To correct half the angle.

$$(1/2)\ \text{Theta} = (0.5(\text{alfa})\ t_2^2)_{\text{control}}$$

or for the total angle

$$\text{Theta} = ((\text{alfa})t_2^2)_{\text{control}}$$

Since this angle must equal the total angle produce by the disturbance, and the correction burn to stop motion.

$$((\text{alfa})t_2^2)_{\text{control}} = (0.5(\text{alfa})\ t^2)_{\text{disturbance}} - (0.5(\text{alfa})t_1^2)_{\text{control}}$$

Solving for t_2, and t_1, and adding these gives t_{total}.

$$t_{total} = t_2 + t_1 \qquad \underline{\text{Equation 1}}$$

Another frequently used equation, is

$$I_{sp} = F\,t/(m_p\,g_o)$$

For hydrazine I_{sp} = 288, with two 0.89 N thruster, F = 1.78 N, g_o = 9.8 m/sec/sec.

Solving for mass of propellant, as a function of total firing time.

$m_p = t_{total}/1586$, in kg <u>Equation 2</u>

Phase 1: Earth Spiral Out

In phase one, the major effects are atmospheric drag, gravity gradient, and internal torques. Since the spacecraft has no extended booms, gravity gradient is not a problem. At the orbit of the craft, aerodynamic drag is negligible, therefore only internal torques are considered. Considering the solar panel tracking torques only, since they are the largest. Using NASA standard .2 pound force thrusters (.89 N).

$I_{yy} = mb^2/12$ = 117 kg(5m)^2/12 = 156 kg m^2

Torque available from AACS = T_{avl} = I(alfa) = (#thrusters)(F)(l) = (2)(.89 N)(6 m) = 10.68 Nm

10.68 Nm = 2 (156)(alfa), since the maximum turning rate is determined by the attitude control system.

(alfa) = 0.034 rad/sec^2

theta = 1/2 (alfa) t^2

If maximum misalignment of the solar panels is pi/16 radians, then the time to turn solar array pi/8 radians (-pi/16 to +pi/16) is given by:

$t_{realign}$ = (2 pi/((8)(.0034)))$^{1/2}$ = 4.8 seconds

Total firing time for the control rockets is:

(#revolutions)(2 pi) = (7875 rev)(2 pi rad/rev) = 49480 rad

theta = 1/2 (alfa) t^2

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$49480 \text{ rad} = 1/2\ (.034 \text{ rad/sec}^2)t^2$

$t = 1706$ sec

Using Equation 2, $m_p = 1.1$ kg

The SEP's torque is apparently capable of counter-acting the above mentioned environmental effect according to mission management.

Phase 2: Earth to Mars Transfer

In phase two, there are no science requirements, the only job will be to overcome the torques from the solar arrays, the antennas, and a possible torque from solar radiation. The calculations below are worst case, so the system will be over-designed rather than under designed. Solar radiation pressure works out as follows:

From a simple calculation of area on each side of the center of gravity is about 80 m^2 vs 110 m^2, or about 30 square meters.

and $l = 4$ m = distance from center of area to center of mass.

$P = F/A = 4.5 \times 10^{-6}\cos(\text{theta})((1-k_s)\cos(\text{theta}) + 0.67k_d)$

where k_s = specular coefficient

and k_d = diffuse coefficient

typical case (Sutton) $k_s = 0.9$, $k_d = 0.5$, theta = 0, for body, and .25, and .01 for solar cells.

$P = 4.5 \times 10^{-6}(0.435) = 1.96 \times 10^{-6}$ N/m^2 for the body, and 1.15×10^{-6} for the solar cells. Where 4.5×10^{-6} , comes from the solar intensity as 1 AU, since this value will decrease as the spacecraft goes out the fuel requirement will be slightly less.

F = PA = 0.0000157 N, and .0000127 N respectively, taking the difference F=.000003N

Fl = T = 0.000012 Nm from sun = I (alfa), where I = 1.35 E 5

alfa from sun = 8.9×10^{-10} rad/sec/sec

$T=Fl= 2(.89)(6) = I(alfa)=1.35 \times 10^5 (alfa)$ for attitude control

alfa = 7.9×10^{-5} rad/sec/sec for control

t = 250 days = 2.16×10^7 sec, for Earth to Mars transfer.

Using Equations 1, and 2.

$t_{attitude\ control}$ = 51500 seconds

m_p = 32 kg

Solar array pointing requirements, as far as turning goes, should be less of a problem than at Earth so an analysis will not be done. Even if the required turning is more, it will probably be periodic, so the CMGs will be able to handle it.

Phase 3: Aerobraking

Phase three, the aerobraking maneuver, will be further divided into to two sub-phases, space flight, and atmospheric flight.

On each revolution of the planet, the space craft must be turned 360 degrees, or 2 pi radians. The time for each orbit is more than a day. Turning time must be a small fraction of this. The torque provided by the attitude control system is:

T=Fl=2 thrusters(.89 N each)(7 m)

=12.46 Nm=I_{CG}(alfa)

with $I_{zz}=1.35 \times 10^5$, alfa = 9.2×10^{-5} rad/sec/sec

In calculating the time for 1 revolution, theta initial = 0, theta final = 2 pi, omega initial = 0, omega final = 0, alfa initial = 9.2×10^{-5} rad/sec/sec, alfa final = -9.2×10^{-5} rad/sec/sec.

Theta = 0.5 (alfa) t^2, but this is assuming a linear angular velocity increase, for the final angular velocity to be zero, half the time must be spent accelerating, and half decelerating, therefore the time is twice as long.

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$t^2 = 2(4\ pi/1.04 \times 10^{-4}) = 492$ sec for total turn around. This is about 8 min., which is only a small fraction of the total orbit time. Also the only half the turning is done in space, the rest is done while in the atmosphere. Since 10 turns are needed, the total firing time is 4920 seconds. Using the Isp formula $m_p = 3.1$ kg

Attitude control in the atmosphere will be the hardest to determine. We are assuming the spacecraft is stable, that is it will tend to keep the aerobrake in front, while in the atmosphere. Therefore a constant torque on the shield is not needed. With a large shield, however, wind gusts on one part of the shield could produce a significant torque. It is impossible to determine how much of a torque, since AEROB does not say how far into the atmosphere the spacecraft goes. Therefore no calculations will be done in that area.

After aerobraking is done, and final orbit is reached, a small delta V burn will be made to move the re-entry vehicle away from the instrument bus. Using a mass of 3000 kg for the reentry vehicle, and an Isp of 288, the fuel requirement works out to be around 2 kg, from delta V = Isp go ln(M1/M2) equation.

Phase 4: Mars Orbit

For the Mars orbit the major effects are, internal, and external torques, and gravity gradients. Assuming the solar arrays turn 2 pi radians per day, the analysis is similar to phase 1, except with smaller arrays. Therefore, since fuel requirements are less than 1 kg, no further calculations will be done with internal torques.

Worst case for the gravity gradient is if the solar array is positioned in such a way as to have part of the array closer to Mars, than the rest of the satellite (see figure 2, page AACS-12). In actuality the solar array rotates in such a way that it is either horizontal with the surface, perpendicular to it, or somewhere in between. It should rarely be perpendicular to it, so in our analysis we are assuming it is 45 degrees from perpendicular with the surface, and that it is in this position 1/3 of the 4 year design life time, or 42 million seconds. $(alfa)_{control} = 2(.89\ N)(3\ m)/(117\ kg/m^2) =$ 0.046 rad/sec/sec. Using a moment arm of 3 meters, and a difference in altitude of 1.23 meters, the calculation is as follows. The formula for gravity at orbit altitude is, $g = g_0((R/(R+h))^2$. Using this formula gravity for the main part of the space craft is 2.845020432 m/sec/sec, g for the solar array is 2.845022235 m/sec/sec. The difference is 1.8×10^{-6} m/sec/sec, over a 1.23 meter difference in altitude. $T = Fl = m(\text{delta } g)l = 1.78 \times 10^{-5}$ Nm. T=I(alfa), therefore alfa from solar torque, with $I = 117\ kgm^2$, is equal to 1.5×10^{-7} rad/sec/sec. Alfa for control is 0.046 rad/sec/sec. Using equation 1, firing time is 53,770 seconds Using equation 2, mass of propellant needed is 34 kg.

Totals

Because of the fact that we are using the SEPs thrusters, attitude control will not have to make a delta V burn to pull away from the space station. Aerobraking places satellite in final orbit, so no delta V is needed for this, unless there is a slight placement error, as long as the delta V is only a few meters per second, the fuel consumption is only a couple of kilograms.

Adding up the requirements for the four phases gives 70 kg. This hydrazine fuel will be placed in two tanks. One on the reentry vehicle, and the other on the instrument bus, each carrying 35 kg. Because the system was over-designed no extra fuel will be added for a safety margin. Hydrazine has a density of 1200 kg/m^3. This makes the 35 kg tanks, 0.029 cubic meters.

Trade Studies

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In chosing between 3-axis-active, spin, or dual spin stabilization methods, Things like instrument pointing, solar panel pointing, and complexity needed to be considered. Dual spin was too complex, and massive, simple spin did not allow for a means of pointing instruments, so the obvious choice was 3-axis-active.

Two systems are capable of providing the needed torques to keep the spacecraft pointing in the right direction. These systems are mass expulsion, and control moment gyros (CMGs). A combination of these is also possible, with an increase in complexity. This is the method we chose, for our spacecraft. It has the main advantage of reducing control propellant, at only a slight increase in system mass. Overall system mass will probably be less, however.

This reduced mass, and increased reliability are important for the space craft. The CMGs also allow for smoother, more precise turning, when needed. They also add stability if a control thruster fails.

In selecting components for the AACS, top priority was given to existing, and proven components. This reduces cost, and increases reliability, because the component has already had the bugs worked out.

Design

Our final attitude control design consists of 26 control thrusters, placed on the spacecraft. 12 are placed on the satellite, 12 on the reentry vehicle, and two on the main propulsion system these thrusters are placed so that they apply moments through the center of gravity, on both the spacecraft, and the satellite. The instrument bus also has a set of control moment gyros, placed at the satellite's center of gravity. These reduce the fuel consumption, resulting from alternating torques, or torques that are applied in one direction, and then the other. The CMGs store the energy during a torque in one direction, the dissipate it when the torque is applied in the other direction (see figures 1&2, page AACS-12).

For information on the attitude control computer, see the computer section.

The individual subsystem sections have included data on the actuators needed in those subsystems, so they are not included here.

Other sensors are all NASA standard. No new technology needed to be developed for the AACS, so proven equipment was selected where possible. See table for more information.

PROBLEM AREAS

One of the main problems in designing the AACS, is a lack of information, on the environment the spacecraft will be traveling in.

Another problem is that certain information, for the spacecraft, was not available in time, for the paper, or the presentation. The work done in the AACS section is based on crude estimates, and educated guesses, for moments of inertia, areas, center of gravity, center of area, and placement of components.

Specific component placement is to done using inert, and has also not been determined yet, therefore only approximate locations can be determined.

SUMMARY OF TRADE STUDIES

Stabilization Method	advantages	disadvantage
3-axis-active (chosen)	-accurate pointing -simple -lower cost than others	-easily perturbed
spin/dual spin	-stable	-increased

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complexity

Control Method	advantages	disadvantages
control rockets	-simple -accurate	-limited lifetime
CMGs	-stable	-needs momentum dumping rockets
Combination of CMGs and rockets (chosen)	-more stable than pure rocket -possibly less massive	-more complex than others -more expensive

Equipment list

ITEM	NUMBER REQUIRED	COMMENTS	REFERENCE
Sun sensor	4	+-70° field of view	Dwyers notes
Control thrusters	26	0.89 N thrust each standard NASA (.2 lbf)	241 notes
Propellant tanks	2	0.029 m^3, and 0.029 m^3	
Radar Altimeter	1	Good resolution at 1000 m	
Control Moment Gyros	cluster of 3	placed at CG of satellite	241 notes
Actuators	handled by individual subsystems		
Star tracker	1	2 axis NASA standard 8kg	241 notes
Gyros	3	3.7 kg	241 notes
Accelerometers	3		
fuel		70 kg, Isp=288	Sutton
Total Mass		110 kg	

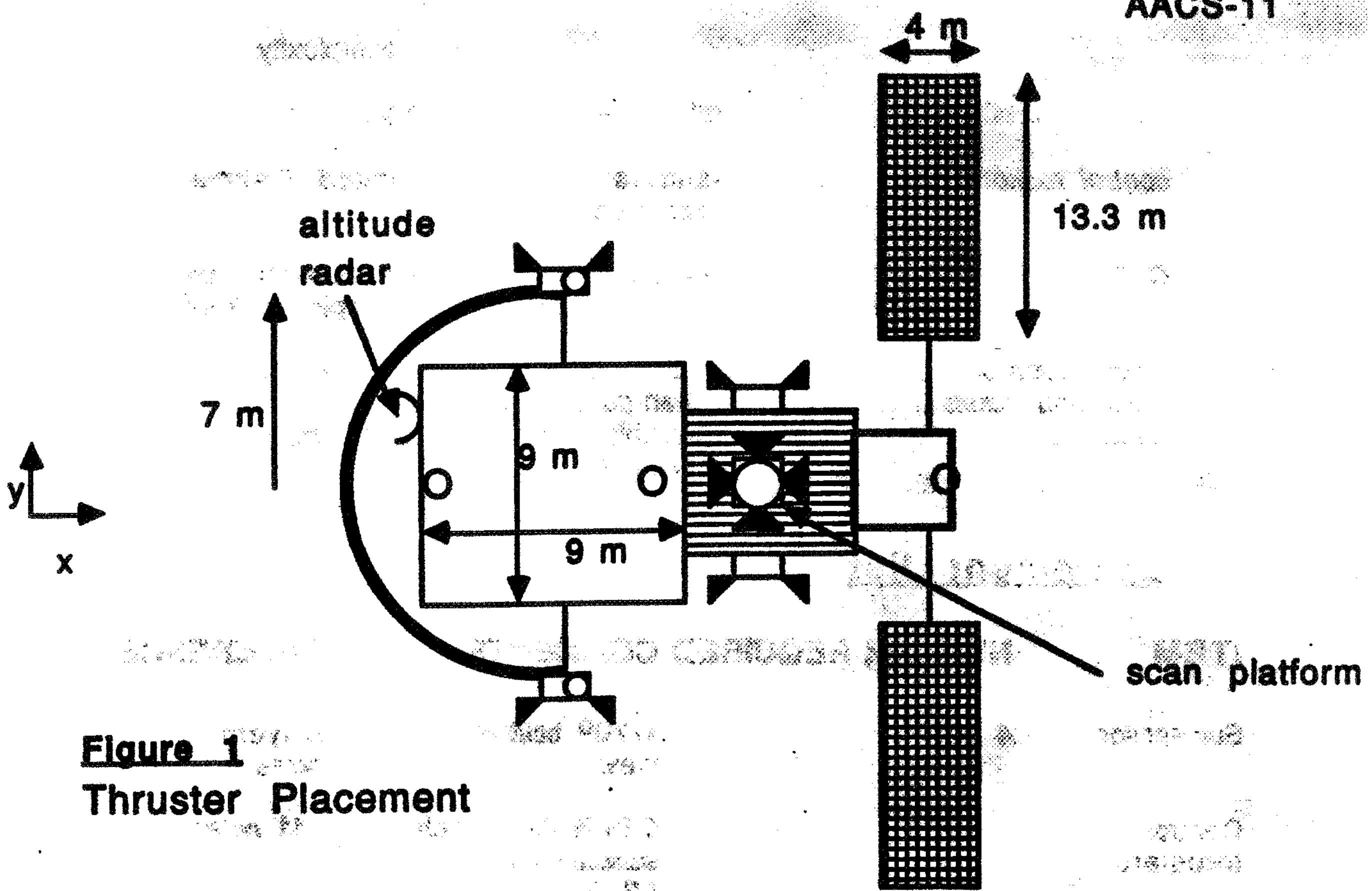

Figure 1
Thruster Placement

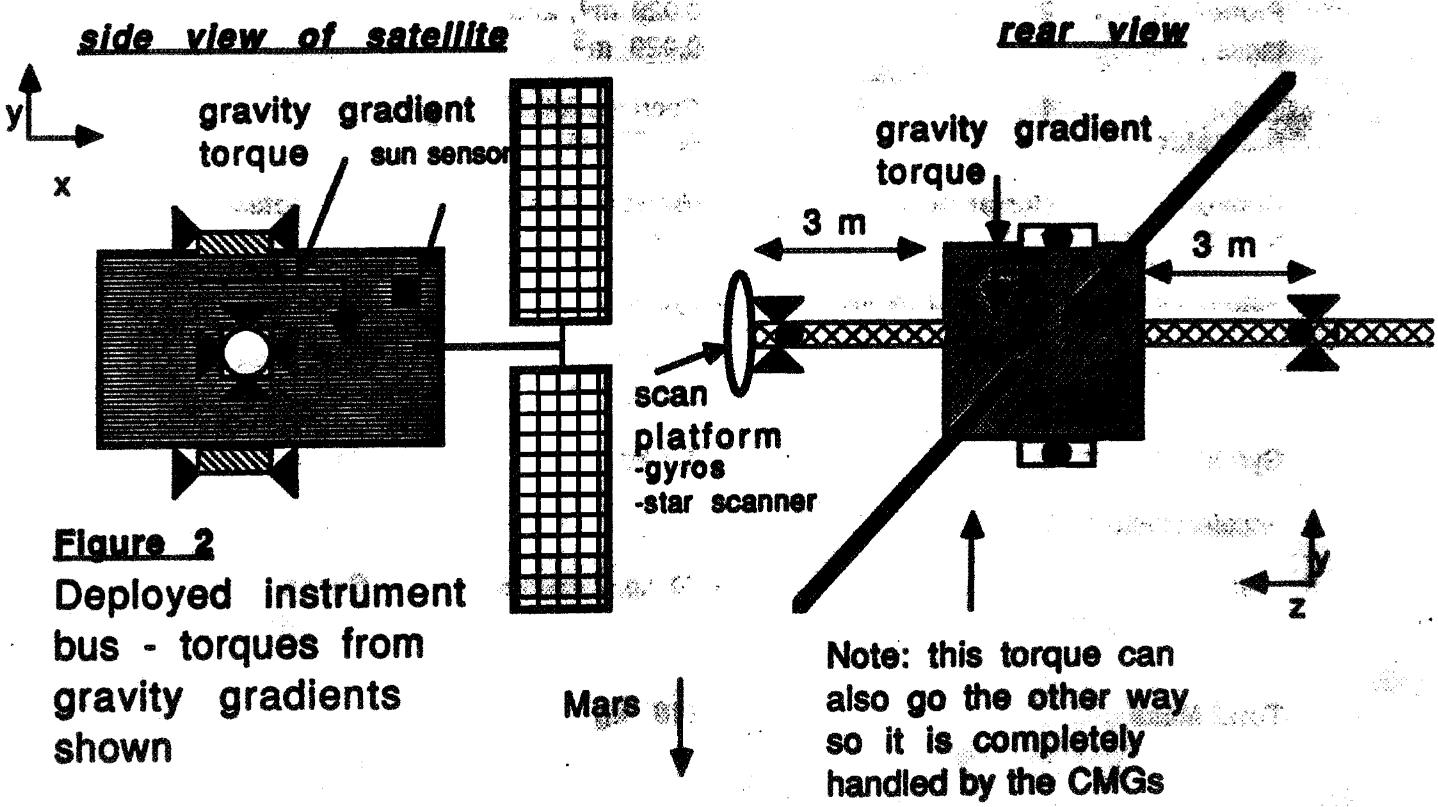

Figure 2
Deployed instrument bus - torques from gravity gradients shown

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REFERENCES

Rocket Propulsion Elements, G. P. Sutton

AAE 241 Class Notes, M. Lembeck

AAE 391 Class Notes, Dwyer

Principles of Dynamics, Greenwood

Mission Science Report:

Investigating Mars; for the most part, the exploration of mars has been limited to imagery and the continuing search for life. The SLAT Mars science platform will for the first time take a detailed look at several aspects of this neighboring planet that will be instrumental in the continuing development of Martian research bases and eventual colonies. Listed bellow are the specific science objectives for this mission.

Determine the elemental and mineralogical character of the Martian surface on a global basis.

Determine the distribution, abundance, sources, and concentrations of volatile materials and dust.

Define the global gravitational field.

Measure the global topography.

Explore the atmospheric structure and its circulation in detail.

Establish the nature of the Martian magnetic field.

Besides for the above listed mission requirements, the science subsystem is responsible for confirming that proper landing conditions are present prior to the re-entry of cargo module.

The design will accomplish these objectives while taking into consideration the following mission guidelines, all of which have been taken as applicable to this subsystem.

1)Off the shelf equipment shall be stresses where available.

2)All equipment must be available prior to 1999.

3)The design shall stress simplicity, reliability, efficient mass, and low cost.

4)Equipment selections shall favor versatility in both mission and deployment.

Research Strategy:

There where 3 major categories, of equipment sources, studied in preparation for this proposal. First to be studied where Earth resource satellites including LandSat, SeaSat,ERS-1,

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and GeoSat. These missions greatly influenced the path in which the research progressed in that the heart of orbital sensor technology is based right here at Earth. In the case of each science objective, an Earth mission has already been performed for a similar purpose. The second category included space probe missions such as the Mariner and Mariner Mark II programs. These programs provided an insight into the types of instrumentation used in modern space exploration. The third and most useful is what we will call the mars science category. This included all prior Martian studies as well as planned mars missions such as NASA's 'Mars Observer' and ESA's 'Kepler Mars Orbiter'. These two missions were of prime importance as they have the same science profile as the mission under study and are both scheduled to fly prior to this missions launch date. In both obiter, a detailed look at each instrument and the corresponding requirement provided several possible mission profiles with which to choose from.

Trade Study:

First we will examine the European Space Agencies Kepler Mars Mission. In Dec(82), ESA publish a 'Phase A' study report on plans for a mission to mars that was to be launched in the mid 1990's. Generated in response to an RFP issued in July of 1980, Kepler is to be a comprehensive Mars science mission with almost the exact same goals that are now in consideration. Though the mission is on hold, the report does explain, in detail, the science and planning phases for ESA's proposed interdisciplinary mars orbiter mission. The Kepler orbiter has many advantages in that it meets the guidelines given in the RFP concerning off the shelf technology, cost, availability, and reliability. The disadvantages of this payload consist of slight differences in science goals.

Next we well examine the Mars Observer Mission. Announced in April of 1985, Mars Observer will be deployed as part of a SSEC strategy for the exploration of the inner solar system. It will be designed to work within an adapted Earth-orbital spacecraft structure that has no allowance for scan platform mounting. A list of the planned payload instruments is provided on page (sci-8) along with the majority of the relevant hardware data. Also provided are two pages from the recent JPL presentation, on Mars Observer, which depict the various instruments and lists the principle investigators for each(page sci-9) as well as the 'Definition of Payload' (page sci-10). The listed equipment will total 123.6kg and require 105.1watts of power which is well within the capabilities of our spacecraft. This payload is nearly ideal for our needs as it was designed under the same science objectives that are now under consideration and is assumed to be deployed prior to the final assembly of this mission, allowing for direct field testing. The only possible deficiency in this profiles use on SLAT is the need for superior atmospheric data to be used in aircraft and ground base support.

Mariner Mark II instrumentation will be the final main element in the trade study. Similar in type to the Kepler equipment, the Mariner program provides a much greater selection from which to choose. The exact abilities of each piece of this equipment were not available in such detail as to make a good comparison with Kepler possible. This forces a less accurate comparison to be made based on power/mass requirements versus data produced. On this basis alone, the Kepler equipment is found slightly superior in that it produces more data for the mass and power required to operate. This in no way should rule out Mariner equipment as the difference is slight; however, being as the Kepler instruments have been specifically organized and configured for a Mars mission, they will receive a higher selection priority.

In each of the three mission sets discussed above, the SLAT science requires could be met without additions from any other source, but other considerations must be made. First, the M.S.O.'s are not the sole responsibility for the science platform, it is also required to provide at least moderate support to the Mars colonies science survey. Specifically the survey work, done by air, is to be supported as much as practically possible. This support, which will be discussed later in this report, should consist of detailed, accurate information on the atmospheric circulation, dust storm tracking, and terrain of Mars. The required accuracies in these key area's may dictate a more sophisticated equipment selection than a simple orbital

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science mission would dictate. In response to this, research was done in the area of space borne topographic and atmospheric flow mapping was done. This research led to synthetic-aperture and atm. sounding radar technology and then to ESA's AMI unit. Planned for use aboard the ERS-1, it is capable of performing both types of work with a accuracy.(see page sci-11)

Tentative Mission Profile:

In the final analysis, the decision of the science division is to pattern the majority of the science payload after the Mars Observer mission. Although the equipment used in the Kepler mission was a refinement of previously developed technology and of less cost, the NASA hardware had the advantages of superior resolution and data quality as well as being developed by 'Home Institutions'. A second consideration in this choice is the fact that Mars Observer will have flown prior to this mission thus balancing Keplers major advantages in prior use as well as the desire for 'off the shelf' hardware selection. Besides for the M.O. equipment, provisions have been made for a C-band Active Microwave Instrument. To be obtained either through the ESA or via home development, the decision whether or not to include AMI on the mission will depend on an environmental estimation of dust storm interference with the mapping spectrograph and accuracy requirements for mapping and atmospheric flow. The superior data resolution and tolerance of atmospheric conditions make the AMI a desirable addition to the base Mars Observer payload.

Problem Area's:

The primary source of trouble is the data volume produced by the science and imaging payload. With an average data generation rate of 100.73 Ms/sec during orbit (73Kbs w/o AMI-SAR running) and a maximum transmission rate to earth of 270Kbs, we are left with periods of heavy data backlog. In response to the problem, data reduction techniques will be employed on the AMI-SAR data(100 Mbs) that will reduce it by a factor of two to three in complete form and by a factor of four to five by data filtering.

Data filtering is the process by which the data is reduced by running it though a pre-processing program, within the on board computer, as it is obtained. This will remove undesired noise and/or unwanted data before it is formatted for transmission. At present, the best solution to the large data rate is for the AMI-SAR data to be filtered into three subsets each to be recorded and transmitted on a different pass over a given area. These subsets would include surface geography, surface integrity, and data involving geologic levels or, in the case of the poles, ice layers depths. Though this process, three maps would be generated for each area which would then be overlaid. This would leave and end rate of approximately 10Mbs from the AMI-SAR system which will be obtained though a preplanned scan-transmit schedule in order to prevent excessive data backlog.

Ground Support:

Despite requests to the aircraft design management, a statement of mission support need has not been received. An estimate of ground coordination has thus been made with the following recommendations for procedural development.

1)Direct instrument calibration to be performed during orbiter fly-by, over the Mars base.

2)Analysis of the local atm, soil, and geography will be compared with orbiter data for accuracy estimates so that a scientific control can be established.

3)Orbiter will provide information on atm currents to the aircraft personal on the surface of Mars.

4)In addition to item3, the orbiter will also provide adv warning of hazardous atm conditions such as dust storms or other conditions which could prove detrimental to ground personal and operations.

5)Communications, telemetry, and data storage and control will be interconnected between the Mars base and orbiter for the purposes of backup storage and long range communications.

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Additional Mission Considerations and Options:

At this point, several inconsistencies in the basic premise of this mission should be addressed. First, and perhaps the most important, is the fact that prior to the establishment of a Mars base, this very mission would be an absolute prerequisite for almost all of the planning stages. Thus the assumption must be made that the Mars Observer or some similar mission would have to have been flown. Through analysis of data acquired for the planning of the Martian base, a better and more specialize set of sensors may be in order for this mission. Based on this situation, the science division has provided a list of alternate science equipment(page sci-11) that would still be applicable to given science requirements, but would provide scientists with a greater diversity in data type for analysis.

Included in this list is the previously presented option of the AMI. It's original addition to the mission was for the expressed purpose of acquiring the more detailed atmospheric and topographic data. It is, however, the opinion of the science divisions that the AMI will be a first round choice in this modified equipment roster and thus it has been designed into this mission in order to lower the cost and development time of the final mission payload.

The next piece of equipment in the alternate list is the x-ray spectrometer. X-ray technology has been used in several prior missions to mars with perhaps the most pertinent example being the Mars 5 which, using a simple x-ray spectrometer, obtained a K/U ratio of 2,000. This value was later revised to 8,000 and thus leaves the results highly in doubt and makes it desirable to include a modern version of this equipment on our mission. It will also provide a rescan of the 10^{-7} to 10^{-13}m spectral band thus covering the the gaps, in the spectral range, left by the Mars Observer Payload.

Lastly, we have an abstract from the instrument list for the Kepler mission that contains the most useful substitutes for the present selections. Included in this are several instruments given within the strawman payload provided in Mars Observers original announcement that have not been scheduled for the mission. Some of this equipment may be of interest to researchers for both second priority experiments and for data conformation by comparative analysis.

This concludes the science devotions report on instrument selection. On page (sci-12) please find a rough working diagram of orbiter complete with tentative payload distributions.

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NASA
Mars Observer

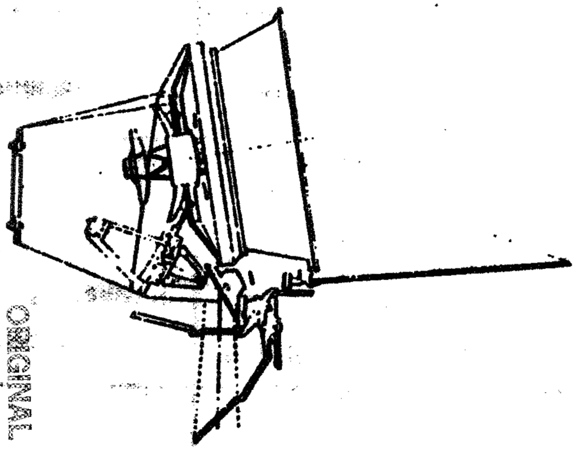
Instrument Selection List

Information provided in JPL Mars Observer Mission presentation(spr'88)

Instrument	Acronym	Mass(kg)/Power(w)	Data Rate bits/sec	Op temp(k)	Derived Parameters
Spectrometers Visual-Infrared mapping	VIMS	22/11	240-30k	87	1) km-res mosaics(320 spec-ch) to indentify mineral and chem units. as well as the dist of surface volatiles 2) regional mapping at 10km res.
Gamma-ray	GRS	25/10.3	600	<100	H,C,O,Na,Mg,Al,Si,Cl,K,Ca,Ti,Cr,Mn,Fe,Ni Gd,Th,U (.2 to 10 MeV)
Thermal Emission	TES	11/15	420-5k		1)Det and map the composition of surface minerals rocks, and ices 2)Study the comp, size, and spac res and temporal distribution of atm dust particles 3)Locate and examine H_2O and CO_2 condensate clouds 4)Study boundry var and energy bal of polar caps 5)Measure thermophysical properties of surface 6)Det atm temp-press var(seasonal),ozone,water var.
Press Mod IR Radiometer	PMIRR	25.7/29	154	<80	1)Map the therm struct vs. time up to 80km alt. 2)Map the atm dust loading variations(global,vert,temp) 3)Map the vert dist of water vapor(seasonal,spacial) 4)Distinguish between atm condensates and map their spatialand temporal variation 5)Map the seasonal and spacial varaility of atm press 6)Monitor polar radiation balance
Magnetometer	MAG	4.4/4	162/648		1)Determine the Martian magnetic field 2)Monitor the Mars/solar wind plama int. 3)Remotely sense ionosphere
Radar Altimeter / Radiometer	RA	24.5/28.3	384		1)Provide topographic height<0.5% of elev change within footprint and RMS slope information. 2)Surface brightness@13.6GHz to within 2.5K 3)Sample radar return waveforms for precise range corrections and the char of surface properties.
Camera	MOC	11/7.5	216-33.7k		Optics for atm &surface changes

SCIENCE INVESTIGATORS AND INSTRUMENTS

MARS OBSERVER SCIENCE



VISUAL AND INFRARED MAPPING SPECTROMETER - L. A. SODERBLOM, USGS, TEAM LEADER

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MAGNETOMETER - M. H. ACUNA, NASA/GSFC, PRINCIPAL INVESTIGATOR

RADIO SCIENCE - G. L. TYLER, STANFORD, TEAM LEADER

INTERDISCIPLINARY SCIENTISTS

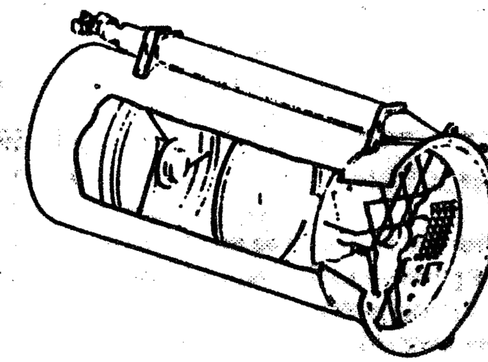
R. E. ARVIDSON, WASHINGTON U - DATA MANAGEMENT AND ARCHIVING

M. H. CARR, USGS - GEOSCIENCES

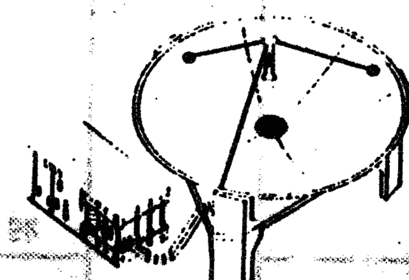
A. P. INGERSOLL, CALTECH - POLAR ATMOSPHERIC SCIENCE

B. M. JAKOSKY, U OF COLORADO - SURFACE/ATMOSPHERE INTERACTIONS

J. B. POLLACK, NASA/AMES RESEARCH CENTER - CLIMATOLOGY

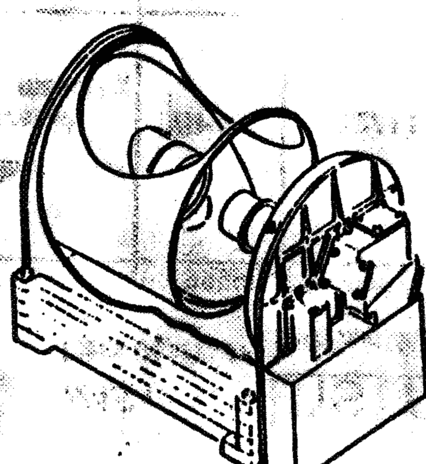


MARS OBSERVER CAMERA - M. MALIN, ARIZONA STATE, PRINCIPAL INVESTIGATOR

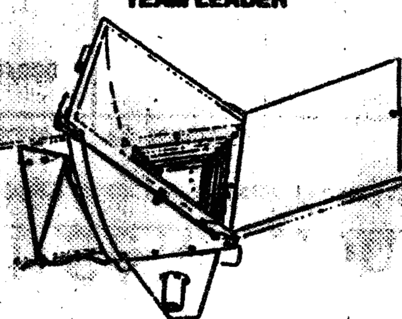


RADAR ALTIMETER/ RADIOMETER - D. E. SMITH, NASA/GSFC, PRINCIPAL INVESTIGATOR

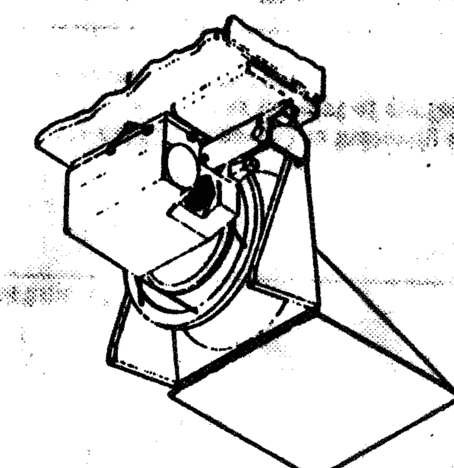
THERMAL EMISSION SPECTROMETER - P. R. CHRISTENSEN, ARIZONA STATE, PRINCIPAL INVESTIGATOR



GAMMA RAY SPECTROMETER - W. V. BOYNTON, U OF ARIZONA, TEAM LEADER



PRESSURE MODULATOR INFRARED RADIOMETER - D. J. McCLEESE, JPL, PRINCIPAL INVESTIGATOR



MARS OBSERVER 1992 MISSION CODE EL BRIEFING

JPL

DEFINITION OF PAYLOAD

- BRIEF INSTRUMENT DESCRIPTIONS
  - FACILITY INSTRUMENTS;
    - VIMS: A MULTI-SPECTRAL GRATING SPECTROMETER COVERING THE BAND FROM 0.4 μ TO 5.2 μ IN 320 SPECTRAL CHANNELS. REQUIRES COOLING OF THE FOCAL PLANE ARRAY TO 87 KELVIN. MASS 22 kg, POWER 11 WATTS
    - GRS: MEASURES GAMMA RAYS FROM 0.2 TO 10 MeV AND THERMAL NEUTRONS SORTED INTO A 8196 CHANNEL PHA. HAS ABILITY TO MEASURE GALACTIC BURSTS. IS MOUNTED ON A SEPARATE 6 m BOOM. THE GERMANIUM CRYSTAL DETECTOR IS COOLED TO < 100 KELVIN. MASS 25 kg, POWER 10.3 WATTS
    - RS: ADDS A USO TO THE S/C TRANSPONDER TO ALLOW TWO WAY LOCK-UP AT OCCULTATION EXIT TO MEASURE ATMOSPHERIC EFFECTS. PRIMARY EXPERIMENT FOR THE MEASUREMENT OF MARS GRAVITY FIELD. MASS 1.6 kg, POWER 2.8 WATTS
  - PI INSTRUMENTS;
    - TES: CONTAINS AN INTERFEROMETER TO MEASURE MINERAL AND ICE COMPOSITION BY THERMAL EMISSION BAND FROM 6.25 TO 50 μ; AND RADIOMETRIC CHANNELS FOR A MEASUREMENT OF BROAD BAND REFLECTANCE (0.3 TO 3.9 μ) AND RADIANCE (0.3 TO 100 μ). MASS 11.7 kg, POWER 15 WATTS AVERAGE
    - MOC: A LINE SCAN CAMERA WITH TWO WIDE ANGLE (GLOBAL COVERAGE- 7.5 km/PIXEL IN 2 COLORS) AND A DUAL RESOLUTION NARROW ANGLE (480 m/PIXEL OR 1.4 m/PIXEL) OPTICS TO STUDY ATMOSPHERIC AND SURFACE FEATURES AND CHANGES. MASS 11.0 kg, POWER 7.5 WATTS AVERAGE
    - PMIRR: A LIMB TO NADIR SCANNING RADIOMETER WITH NINE SPECTRAL CHANNELS——5 FILTER CHANNELS AND TWO PRESSURE MODULATED CELL CHANNELS (CO_2 AND H_2O) AND TWO UNMODULATED CHANNELS AT THE SAME FREQUENCIES. PROVIDES SEASONAL MAPPING OF ATMOSPHERE, CO_2, H_2O AND DUST. REQUIRES DETECTORS TO BE COOLED TO < 80° K, MASS 25.7 kg, POWER 29 WATTS AVERAGE
    - MAG: DUAL TRIAXIAL FLUXGATE MAGNETOMETERS AND ELECTRON REFLECTOMETER SPACED AT 2 INTERVALS ALONG A 6 m BOOM. TO ESTABLISH THE NATURE OF THE GLOBAL MAGNETIC FIELD AND SEARCH FOR CRUSTAL ANOMALIES ON THE SURFACE AT THE RATE OF 2 TO 16 VECTORS/sec. MASS 4.4 kg, POWER 4 WATTS
    - RA: A Ku-BAND (13.6 GHz) RADAR ALTIMETER AND RADIOMETER TO MEASURE SURFACE ELEVATIONS TO < 0.5% AND SURFACE BRIGHTNESS TEMPERATURES AT 13.6 GHz TO A PRECISION OF 2.5 degrees KELVIN. EMPLOYS A 1.0 meter, NADIR POINTED ANTENNA. MASS 24.5 kg, POWER 28.3 WATTS

ESA ERS-1 Kepler

NASA MarinerII

Alternate Selection List

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Instrument	Acronym	Mass(kg)/Power(w)	Data Rate	Derived Parameters
			Mb/sec	
Active Microwave Insrument Package	AMI		100	a)Synthetic Ap Radar(50m special res) b)wave spectromter c)wind scatterometer (wind speed 4-24m/s 10%) (50km special res) (direction 0-360deg 20deg)
Spectrometers			Mb/day	
Neutral Mass	NMS	7/5	1.6	Upper atmosphere density, composition (chemical and isotopic), temperature and eddy coefficient
Ion Mass	IMS	4/1.5	1.6	Ion density and composition
Ultaviolet Visible Mapper	UVS	6/4	1.6	Atmospheric composition, temp from atmospheric light scattering and emissions; surface and dust albedo vs wavelength
Magnetometer	MAG	2.5/2.5	0.8	Magnetic field vector
Radar Altimeter	RAD	12/7	1.6	Topography
Infrared Sounder	IRS	8/5	2.4	Atmospheric temp and humidity, dust opacity and CO_2 partial pressure surface temp and spec reflectance property
Doppler Ranging	GRA			Gravity field
Retarding Potential Analyzer	RPA	3/2	0.8	Ion density and composition
Electron Langmuir Probe Plasma Wave Analyzer	ELP/PWA	3.5/3	1.6	Electron density and temperature wave energy spectra
X-Ray Spectrometer	XRS	14/4	300bps	

SPLAT Science/Support Platform

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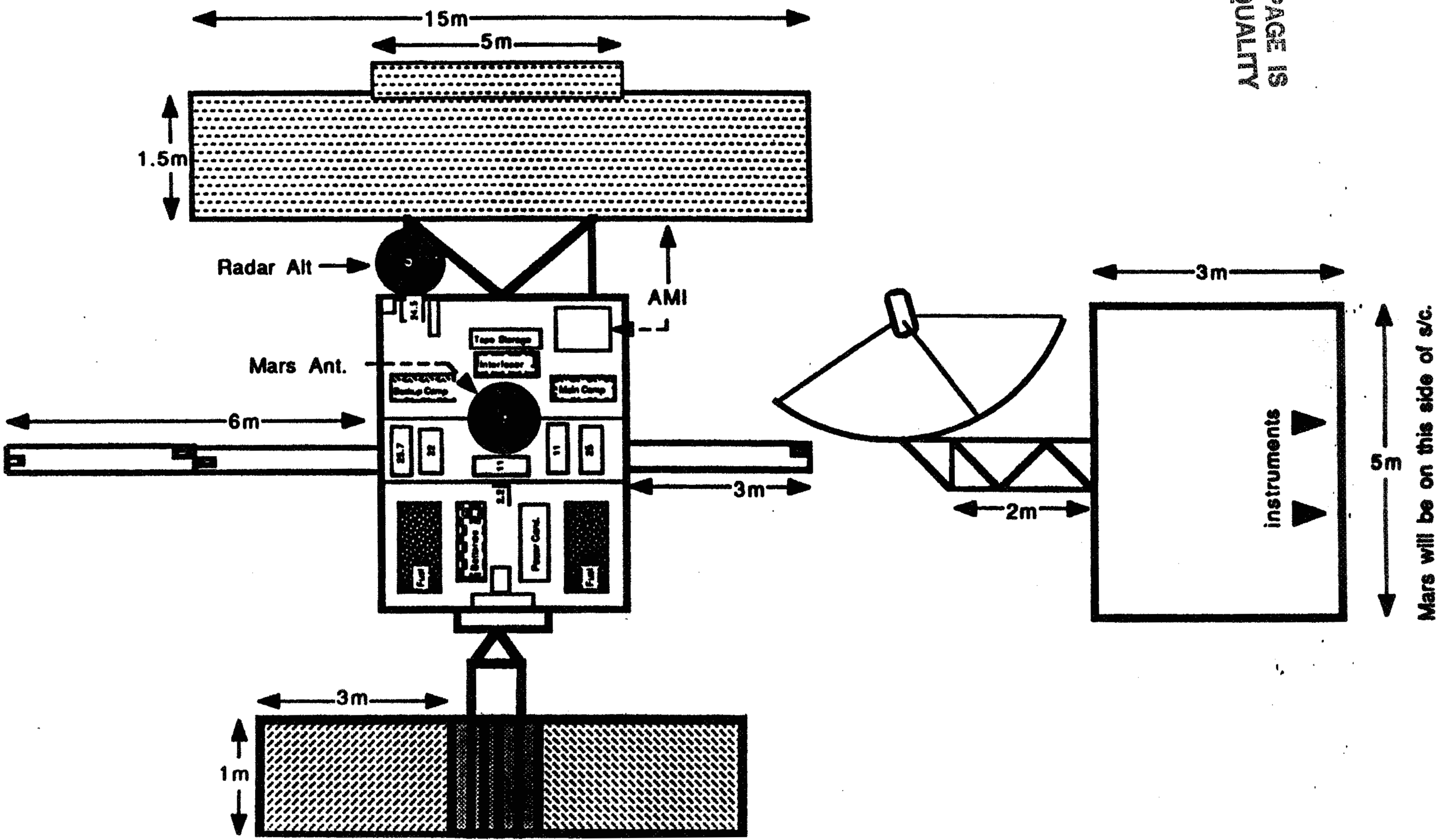

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POWER AND PROPULSION SUBSYSTEM

In designing the power and propulsion subsystem (PPS), certain mission requirements had to be followed. These included use of off-the shelf hardware, no use of materials or techniques expected available after 1998, and a simple, reliable, low cost and low mass design.

To accommodate these requirements, other subsystem requirements were created. First, the subsystem must have a capability to interact with other spacecraft subsystems. These interactions include providing power for the AACS, CDC, and science subsystems, accepting commands from the CDC subsystem, and sending telemetry to the CDC. The PPS subsystem must also be capable of sensoring inputs, such as temperature data, and controlling outputs, such as powering relays. Finally, the subsystem must be self-powered.

A method of attack was needed in order to help develop a PPS design that met the mission and derived requirements. First, the PPS subsystem was divided into its two major sections, propulsion and power. Next, research was done in each section to identify and collect data on different options that could be used. Then, important areas of the options were identified, ranked in importance, and given a rating value that was incorporated into a decision selection equation. The best option in each section was then chosen.

For the propulsion section, the options chosen were put into two subsections, corresponding to stages. The first stage is composed of the propulsion option used to get the spacecraft from Earth to Mars. The second stage is composed of the propulsion option used in aerobraking and circularization maneuvers at Mars. Each identified fuel types and chose the best, identified fuel requirements for the stage, sized tanks, identified and chose pumping systems, sized engines, and finally developed a stage configuration.

For the power section, power requirements from other spacecraft subsections were calculated before the options were subjected to trade studies. The best option for the section was then chosen. Selection and sizing processes were undertaken. Finally a section configuration was developed.

At last, a total subsystem configuration was developed, and problem areas were identified.

PROPULSION- In the propulsion section, 3 different options were compared, including chemical, nuclear, and solar electric propulsion. Data was gathered on each system in important areas, such as mass, cost, safety concerns and reliability. These areas were ranked in order of mission importance, and given a rating to develop a decision equation reflecting their importance. Each option was rated in each area. This was done by giving 100% of the area rating to the system that was the best in that area, 75% to the second best, and 50% to the worst. The ratings for each option's areas were added up, and the option with the highest total rating was selected.(See Chart 1)

The solar electric propulsion system (SEPS) was chosen to be the main propulsion system (Stage 1), and the chemical propulsion system was selected as the secondary propulsion system (Stage 2)

STAGE 1- The SEPS system was chosen to be the main propulsive system because of its mass advantage over other systems (1/4 of the next least massive system), moderate costs, long component lifetime, no outstanding safety concerns in its use, minimal required shuttle support, and short low earth orbit (LEO) assembly time.

The choice of fuel was done by comparing fuels that had been tested with the SEPS system. These included mercury, cesium, and xenon fuels. Xenon was selected because of its higher specific impulse values than mercury at equal power inputs, and the fact that

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its use does not cause handling and safety concerns, as does the use of mercury and cesium.[2]

Once the fuel was chosen, a specific amount of fuel needed to get the spacecraft to Mars was determined by the mission planner. This amount was determined to be 1393 kg. The tankage factor for xenon fuel was found to be higher than that of a chemical fuel (.18 to .06). With these numbers, a tank mass could be calculated. This value is 250.8 kg.

A choice of pumping systems was simplified by finding that the cryogenic system needed for xenon would not improve tank fuel mass capability enough to off-set the mass of the cryogenics. This then effectively eliminated a turbopumping system from consideration. A pressurized fuel feed system was then chosen.(See Fig.1). This system would be much less massive than a cryogenic-turbopump feed system, and has been used frequently with xenon ion propulsion tests.

Engine data was then gathered, which included different sizes and arrangements of ion engines. Each was compared, and a 4-engine module with 30 cm ion engines was selected.[2] This engine arrangement has the advantage of using less components, hence less massive, than comparable Bi-Mod (2-engine) systems. The 30 cm engines have been researched and tested more thoroughly than any others, and have been used with the cluster module.

Each engine with gimbal has a mass of 18 kg, a design power input of 8 kW, and a projected thrust/power ratio of 63 mN/kW, which is a 29% increase from the todays maximum ratio.[3]

The engine module requires 2 neutralizers, 4 engine power processing units (PPUs), 2 neutralizer PPUs, and 3 power supplies per engine and neutralizer PPU. The total mass of this module, excluding engines and gimbals is 180 kg.

In order to reduce spiral out time and minimize subsystem mass, a thrust of 2 N is required. This means that 32 kW of direct line power is needed to accomplish this. The method of power generation will be discussed later.

The main propulsion system, though, can't provide the amount of thrust needed to perform aerobrake maneuvers at Mars. A second, high thrust propulsion system is needed to do this.

STAGE 2- To achieve this thrust, a chemical propulsion system was selected. A liquid chemical system is preferred due to its throttling and restart abilities.

Fuel selection for this stage was accomplished by comparing fuel types, in terms of their properties and characteristics. These fuels included liquid hydrogen, RP-1, hydrazine, and unsymmetrical hydrazine (UDMH). UDMH was chosen for its high specific gravity (.611 g/cubic cm), stability, low freezing and high boiling points, and its prior use in similar space applications.[4] Also, it does not require heavy cryogenics to keep it in liquid form.

Next, an oxidizer was picked. Among those examined were liquid oxygen and fluorine, hydrogen peroxide, and nitrogen tetroxide. Due to its high specific gravity (1.447 g/cubic cm), long storage capability, hypergolic characteristics, and prior use with UDMH in space systems, nitrogen tetroxide was chosen.[4]

With a fuel selected, an amount of fuel necessary to accomplish aerobraking and orbit circularization maneuvers could be calculated. A fuel/oxidizer mass of 1100 kg was needed. Tankage factor for the fuel/oxidizer combination are slightly higher than those of other chemical systems, due to the fact that the nitrogen tetroxide must be kept in a high strength tank due to its high vapor pressure.This tankage factor is .08, which results in a tank mass of 88 kg.

In order to reduce mass, a pressurized gas pumping system was selected.(See Fig 2). It is a more reliable, lower cost, and much less massive system than the use of

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turbopumps. Also, it has been successfully used in space with the fuel/oxidizer combination chosen.

Finally, an engine was needed for the second stage. Research was done to find engines of minimal size that had performed similar functions to aerobraking. A lunar module descent engine was selected because of its small mass (181 kg), proven performance with the fuel/oxidizer/pumping system selected, long rated lifetime (800 sec), high thrust, good throttling capability (10:1), and its use in similar maneuvers.[4]

Engine specifications	
Length	2.3 m
Width (at nozzle)	1.5 m
Fuel mixture ratio	1:1
Specific impulse	300 sec

STAGE 1 CONFIGURATION

Fuel mass	1393 kg
Tank mass	250.8 kg
Fuel Isp	4000 sec
Specific grav. (xenon)	2.41 g/cubic cm [5]
PPU & Neutral. mass	180 kg
Fuel feed system mass	50 kg
No. of fuel tanks	6
Spherical tank diameter	57 cm
Mass of each tank	41.8 kg
Mass of fuel in tank	232.2 kg
Engine mass	72 kg
Stage mass	1955.8 kg

STAGE 2 CONFIGURATION

Fuel mass	1100 kg
Tank mass	88 kg
Fuel/ox specific grav	1.18 g/cubic cm
Engine mass	181 kg
No of tanks	2
Cylindrical tank size	82 m * .6 m
Mass of each tank	44 kg
Mass of fuel in tank	550 kg
Fuel feed system mass	25 kg
Stage mass	1394 kg

POWER- To begin the power section, the power requirements from every subsystem had to be known. The subsystems that required power were the CDC,AACS and science systems. The maximum required power for these was 1.8 kW and a housekeeping value of .15 kW. The propulsion section itself required 32 kW of direct-line power. From these figures and developed decision equation, a power source could be chosen.

Four different options were considered, including RTGs, regenerative fuel cells (RFCs), nuclear reactor, and solar cells. The solar cells were chosen because of their ability to generate large amounts of power, low cost, large availability, no safety concerns in their use, and proven effectiveness in space use. Because of the eclipse periods, batteries were also needed.(See Chart 2)

Types of solar cells, including silicon and gallium arsenide (GaAs) cells, were compared. Cell efficiency, power/area ratio, cost, and availability were used as determining factors. A GaAs cell, with an AlGaAs/GaAs structure was selected.[6] It has a lower cost and larger availability than silicon cells, and a larger power/area ratio than other GaAs cells.

Battery types were compared next. These types included lead-acid, nickel-hydrogen, nickel-cadmium, and lithium-hydrogen batteries. Determining factors were mass, cost, reliability, availability, proven effectiveness, and stored energy density capabilities. Nickel-cadmium batteries were selected. They are much less massive than lead-acid types, much more available than nickel and lithium hydrogen types, and have a superior proven effectiveness in space over all of the others.

Due to the staged nature of the spacecraft, 2 sets of solar arrays were designed. The main and larger solar array was attached to the SEPS stage, providing direct power to the ion engines. It also provided a small amount of power (.15 kW) as a housekeeping allowance to the other subsystems. The second and smaller array will be located next to

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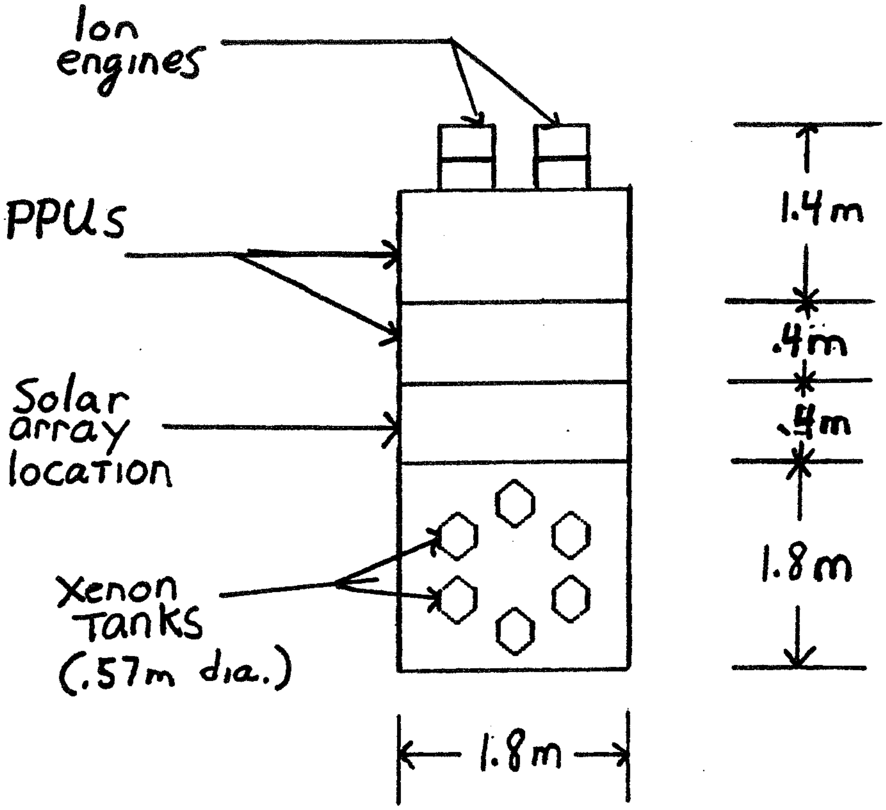
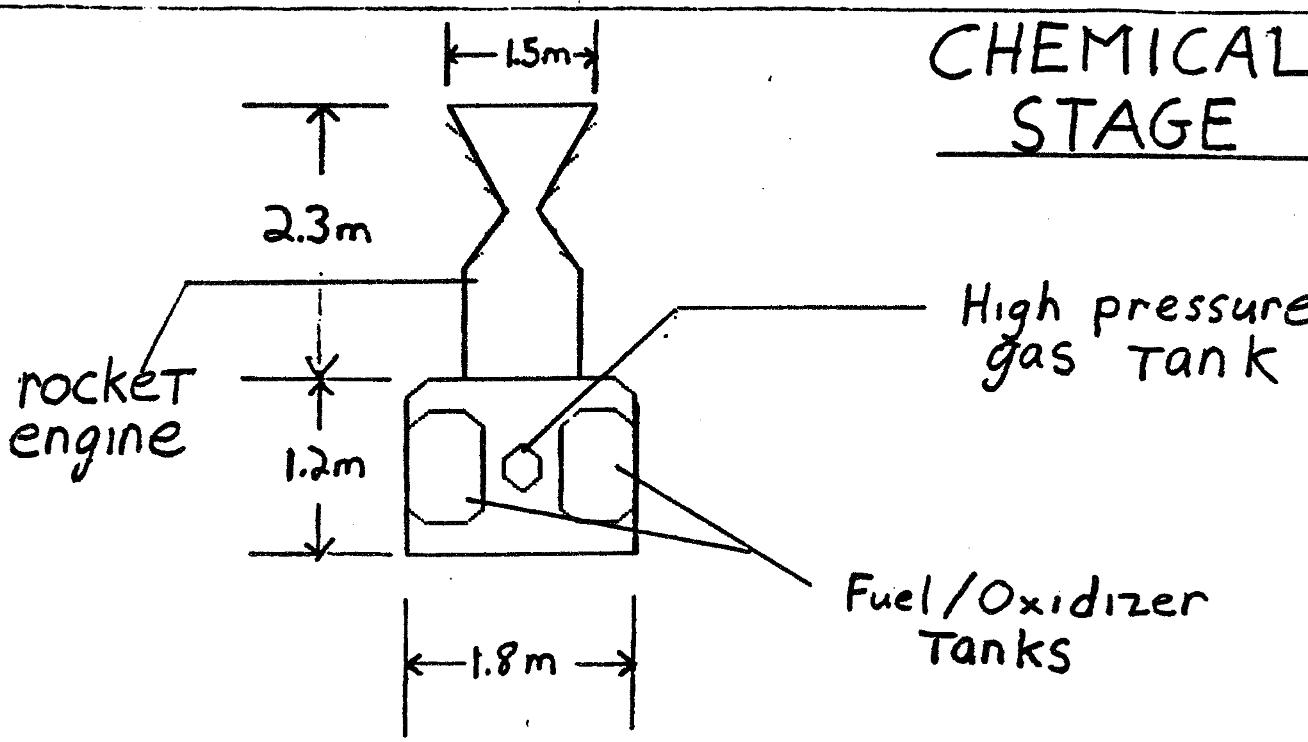
SEPS STAGE
(STAGE 1)



Use of explosive bolts to separate stage.

CHEMICAL STAGE



STAGE 2

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the CDC and science subsystems and provide power to those. It will not be deployed until separation of the second stage. This will be covered in the science subsection report.

The main array was designed to provide 32 kW of power to the ion engines. A sizing allowance was used to compensate for cell degradation. The array is to be constructed in LEO and is to be rigid. The structure was designed this way because there is no need for the array to be retractable; it is to be discarded along with the SEPS stage.

The second array was designed to meet the power demands of the CDC, AACS, and science subsystems. It will be able to provide 1.8 kW of power at Mars. The construction and storage of the array will be discussed in the science subsystem report.

Chart 3 contains array sizing equations, along with cell and array data. The power/area ratio takes into account a .88 packing factor, appropriate solar constant, cell efficiency, temperature degradation factor and operating temperature, along with a projected 40% increase due to technological advances in the area. Chart 4 contains battery sizing equations, along with nickel-cadmium battery data

<u>MAIN SOLAR ARRAY CONFIGURATION</u>- Due to the fact that shadows will be caused by the aerobrake, the main solar array arms were located on the end of booms 6.1 m in length to avoid this problem.

<u>OVERALL CONFIGURATION</u>

<u>SEPS STAGE (STAGE 1)</u>

Length	4 m
Width	1.8 m
Height	1.8 m

<u>CHEMICAL STAGE (STAGE 2)</u>

Length	3.5 m
Width	1.8 m
Height	1.8 m

<u>LARGE SOLAR ARRAY (including boom)</u>

Length of arm	19.4 m
Width of boom	.4 m

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Width of cells 4 m

PROBLEM AREAS- Since projected data was used in sizing of components, reasonable technological advances must be made in these areas for the design masses and sizes to be accurate.

Also, the ion engines cant be operated during eclipse times, since batteries are not provided and would not be cost or mass efficient. This could lead to longer trip times than expected.

Finally, assembly in LEO is required. Advances must be made in the area of space construction in order to provide the necessary support.

FINALIZED TRADE STUDY DECISIONS-The SEPS system was chosen over the chemical and nuclear propulsion systems because of its low mass, among other things. Xenon fuel was chosen over mercury and cesium fuels because of its higher specific impulse and safety in use. For the second stage, liquid chemical propulsion was chosen over solid chemical because of its throttling and restart capabilities. UDMH was chosen over liquid hydrogen, RP-1, and hydrazine because of its high specific gravity, low freezing and high boiling points. Nitrogen tetroxide was selected over liquid oxygen and fluorine, and hydrogen peroxide because of its high specific gravity, and hypergolic characteristics. A pressurized gas fuel pumping system was selected over turbopumps because of its low mass ans higher reliability. Solar cells were chosen over RTGs, RFCs,and nuclear reactors because of low cost, large availability, and no safety concerns in its use. Finally, GaAs cells were chosen over Silicon cells because of their large availability and Ni-Cd batteries were chosen over lead-acid, nickel-and lithium-hydrogen batteries because of their proven effectiveness in space applications.

Figure 1

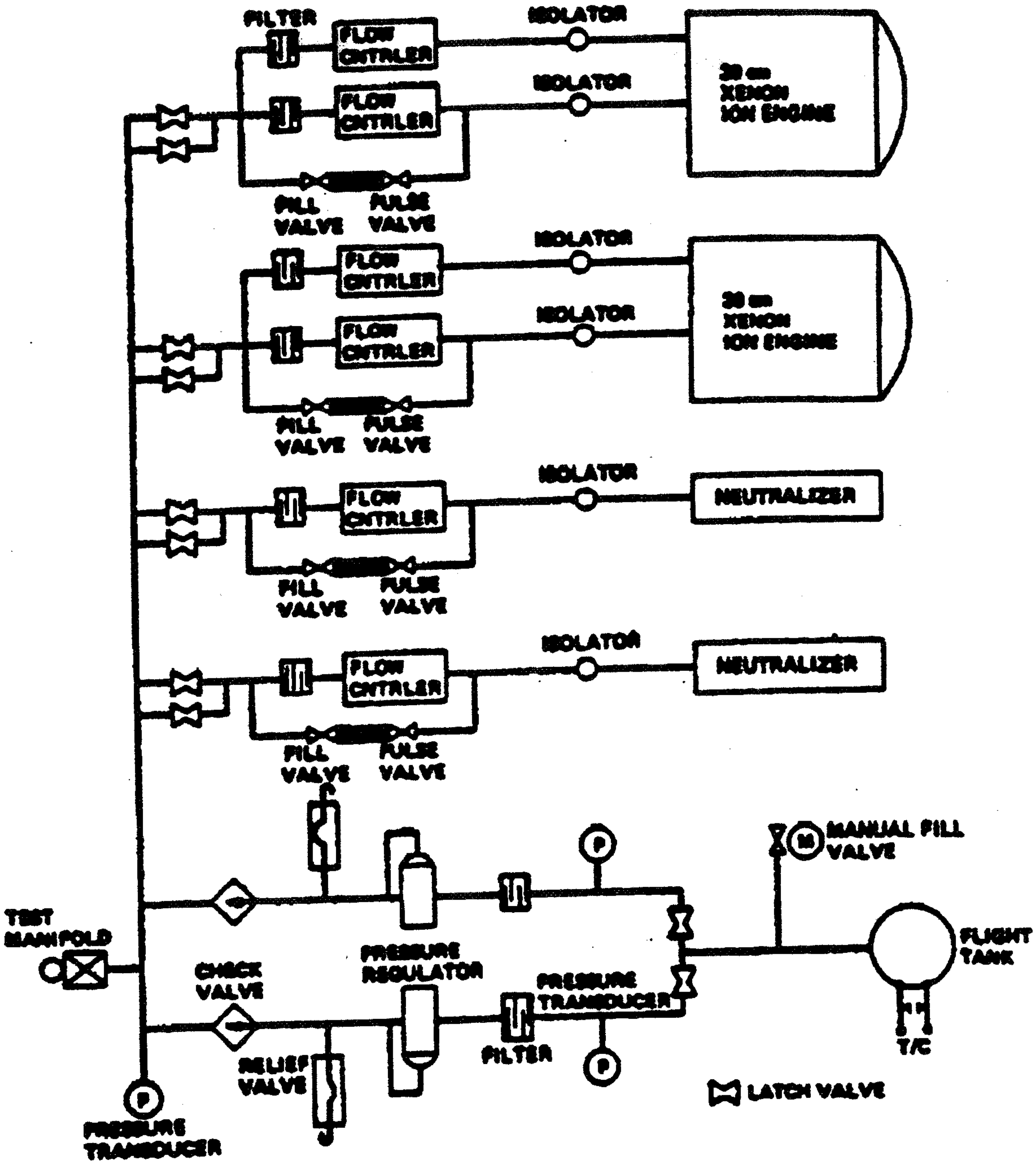

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Figure 2

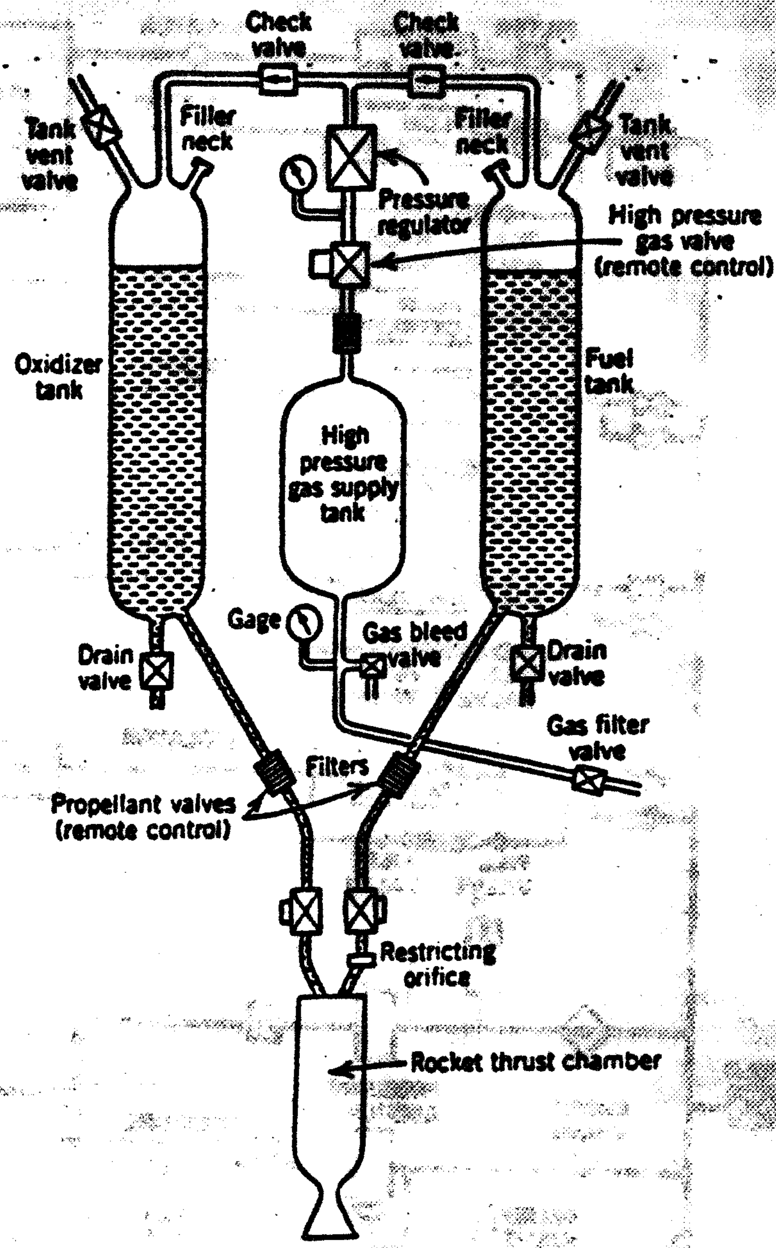

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CHART 1

DECISION EQUATION AND OPTION SELECTION (PROPULSION)

Total mass	.22	Option Best in Category-	100%
Cost	.20	Second Best in Category-	75%
Safety concerns	.18	Third Best in Category-	50%
Reliability	.15		
Component lifetime	.10		
Shuttle support	.08		
LEO assembly time	.05		
Travel time	.02		
	1.0		

	CHEMICAL		NUCLEAR		SEPS *	
Total mass	30000 kg	(.165)	65000 kg	(.110)	7000 kg	(.22)
Cost	Moderate	(.20)	Large	(.10)	Moderate	(.20)
Safety concerns	Some	(.135)	Many	(.09)	None	(.18)
Reliability	Good	(.15)	Good	(.15)	Good	(.15)
Component lifetime[1]	500 hrs	(.05)	1500 hrs	(.075)	10000 hrs	(.10)
Shuttle support(trips)	1 - 2	(.08)	2 - 3	(.06)	1	(.08)
LEO assembly time	Average	(..038)	Long	(.025)	Short	(.05)
Travel time	8-10 mths	(.02)	6-8 mths	(.02)	20 mths	(.01)
Total		(.84)		(.63)		(.99)

"*" denotes chosen option

CHART 2

DECISION EQUATION AND OPTION SELECTION (POWER)

Mass	.25
Cost	.20
Safety concerns	.18
Power generated	.18
Reliability	.12
Availability	.07
	1.0

Option best in category-	100%
Second best in category-	75%
Third best in category-	50%

	RTG		RFC		Nuclear		Solar cells *	
Mass	Small	(.25)	Small	(.25)	Large	(.125)	Avg.	(.188)
Cost	Avg.	(.15)	Avg.	(.15)	Large	(.10)	Small	(.20)
Safety concerns	Few	(.135)	Few	(.135)	Many	(.09)	None	(.18)
Power generated	Small	(.09)	Avg.	(.135)	Large	(.18)	Avg.	(.135)
Reliability	Good	(.12)	Fair	(.09)	Good	(.12)	Good	(.12)
Availability	Ltd.	(.035)	Ltd.	(.035)	Avg.	(.053)	Lrg.	(.07)
Total		(.78)		(.795)		(.668)		(.893)

"*" denotes selected option

Avg.-Average Ltd.-Limited Lrg.-Large

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CHART 3

ARRAY SIZING EQUATIONS AND SPECIFICATIONS

[7] Array power (BOL)= Array power (EOL)/ Time degradation
Array area= Array power (BOL)/ (Cell power/ area)
Array mass= Array power (BOL)/ (Cell power/ mass)
Time degradation= 1- Degradation rate

GaAs cell data

	(projected) power/ area.	.333 kW/ sq. m
[8]	(projected) power/ mass.	.150 kW/ kg
	(assumed) Degradation rate	6% per year
	Time degradation	.94

Large Array

power (BOL)	34 kW
area	102.2 sq. m
no. of array arms	2
mass	227 kg
dimension of array arms	13.3 m * 4 m
harness, sensor, and actuator mass	30 kg
total array mass	257 kg

Small Array

power (BOL)	2 kW
area	6 sq. m
mass	13.5 kg
harness, sensor, and actuator mass	12.5 kg
total array mass	26 kg

TOTAL MASS (LARGE AND SMALL ARRAY) 283 kg

BOL-Beginning of life EOL-End of life

CHART 4

BATTERY SIZING EQUATIONS AND SPECIFICATIONS

[7] Stored energy= Power Load * Eclipse time/ DOD
Battery mass= Stored energy/ Stored energy density
NO. of cells= Stored energy/(Stored energy/ cell)

Nickel-cadmium battery data(Ni-Cd)

Power load	2 kW
Eclipse time	.25 hours
Depth of Discharge (DOD)	.55
Stored energy per cell	31 W-hours
Stored energy density	35 W-hours/ kg

Sizing

Stored energy	.91 kW-hours
Battery mass	26 kg
No. of cells	30
Back-up battery mass	13 kg
No. of cells	15
TOTAL BATTERY MASS	**39 kg**

PROBLEM AREAS

1 Due to use of projected data in sizing components, reasonable technological advances must be made.

2 Ion engines can't be run during eclipse time. Longer trip times may result.

3 LEO assembly is required. Advances must be made in the area of space construction in order to provide the support.

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Command and Data Control Subsystem (CDC)

Introduction

As specified in "Request for Proposal and Preliminary Design of a Manned Mars Aircraft Space Delivery System" (RFP), the Command and Data Control subsystem (CDC) has been defined. The CDC provides data and command links between Earth and the spacecraft. To control the data and commands, the CDC encompass the spacecraft's computers, and in that capacity, controls the spacecraft. On board computers will provide data and command acquisition, processing, storage, and relay. In this report, the method of attack used to fulfill the RFP requirements will be discussed, followed by the RFP requirements relating to the CDC. The trade studies used to select CDC components will be covered, followed by a look at problem areas. Finally, the integrated subsystem will be examined in detail in a system overview.

Method of Attack

This is the procedural algorithm used to achieve the goals set forth by the RFP. The steps taken are:

1. Distill requirements from the RFP and class notes, noting CDC general and specific requirements.

2. Define a set of working specifications to describe the subsystem and it's actions.

3. Collect options to fulfill system requirements and specifications.

4. Conduct trade studies to optimize the subsystem configuration.

5. Determine if the selections made in 4 fulfill the requirements and specifications. If not, find more options or rewrite subsystem specifications, or make less optimal selections.

6. Integrate the subsystem.

7. Derive an implementaton plan.

8. Write the final report.

9. Prepare the corresponding presentation.

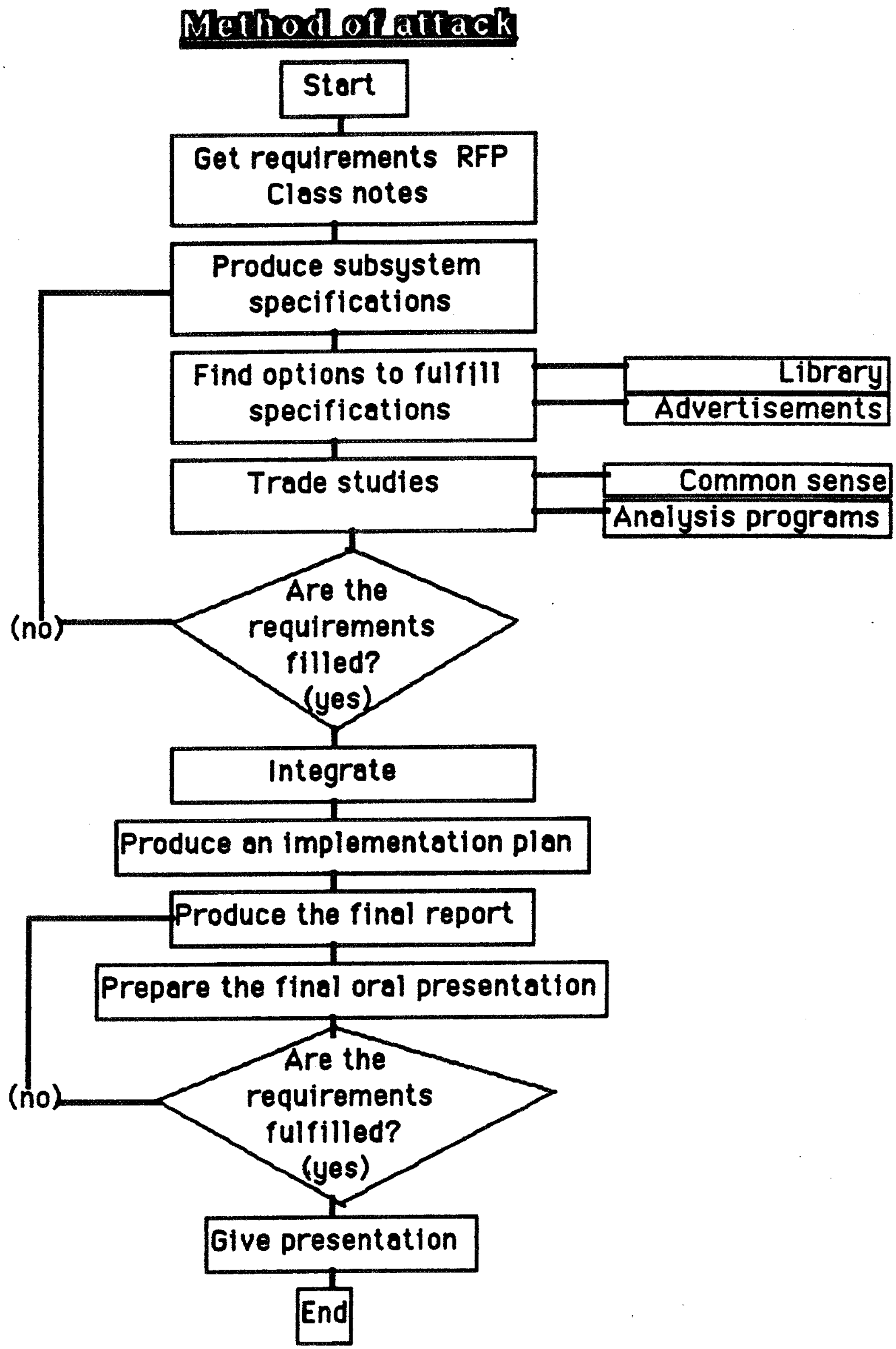
Method of attack
Start
Get requirements RFP
Class notes
Produce subsystem specifications
Find options to fulfill specifications
Library
Advertisements
Trade studies
Common sense
Analysis programs
Are the requirements filled?
(no)
(yes)
Integrate
Produce an implementation plan
Produce the final report
Prepare the final oral presentation
Are the requirements fulfilled?
(no)
(yes)
Give presentation
End

Fig. CDC-1

CDC Requirements

In the RFP, many requirements were put forth . A number of requirements apply to the CDC subsystem, both generally, and specifically.

The general requirements are:

1. The project objective is to develop a conceptual design for the spacecraft system required to deliver an aircraft to the Martian surface. (The key words here are "conceptual design").

2. The spacecraft's components will be delivered to orbit by the Space Shuttle, and will be assembled on-orbit. (First, everything must fit within the Shuttle bay, and second, it must be constructed for easy on-orbit assembly, if any is needed).

3. Nothing in the spacecraft's design should preclude it from performing several possible missions, carrying vastly different payloads to different destinations. (Do not limit the spacecraft to just this mission).

4. It should use off-the-shelf hardware where available. (Do not invent new technologies where existing technologies will fulfill the requirements).

5. It should not use materials or techniques available after 1998.

6. The spacecraft will have a design lifetime of four years but nothing in it's design should preclude it from exceeding this lifetime. (The design lifetime will be assumed to begin at Mars arrival.)

7. The design will stress simplicity, reliability, minimum mass, and low cost. (Keep it simple, inexpensive, and reliable).

The specific requirements are:

1. For the purpose of system integration, the Command and Data Control subsystem is identified.

2. The vehicle will use the latest advances in artificial intelligence (AI) where applicable to enhance mission reliability and reduce mission costs.

3. After the payload has been delivered to the surface, the rest of the spacecraft will act as a relay satellite to support the aircraft.

The requirements put forth in the class notes are:

1. Collect telemetry from the subsystems.

2. Send telemetry to the ground.

3. Send commands to the subsystems.

4. Send power switching commands for the subsystems.

5. Receive power from the Power and Propulsion Subsystem (PPS).

6. Control outputs:

   a. Power switching.

   b. Telemetry to the ground.

   c. Commands.

7. Receive inputs:

   a. Telemetry.

   b. Power requests.

   c. Commands from the ground.

8. Components:

   a. Computer.

   b. Radio.

   c. Antenna.

In order to complete these requirements, and to integrate the CDC with the rest of the spacecraft, further specifications were needed.

The CDC specifications are:

1. The CDC is responsible for all on board computers. This does not cover special control logic needed for the control of specific equipment.

2. The CDC is responsible for Earth-spacecraft and Mars-spacecraft communications.

3. The CDC will employ AI software to enhance mission reliability through early fault detection and correction.

4. Because communications are not possible at all times, the CDC will be able to store data for transmission at a future time.

5. Because of large data quantities supplied by the Science subsystem, the CDC will be able to communicate at the highest reasonable data rate possible.

6. Because of the AI software demands, a highly capable computer will be employed on board.

7. Because of large AI program sizes, and corresponding data, a large data store will be needed.

8. The CDC must be fault tolerant.

With these specifications in hand, options were located, and trade studies were conducted.

Trade studies

For the purpose of clarity, trade studies have been broken down into two categories. Communication equipment will be looked at first, followed by computer equipment.

Trade studies: communications

For the Earth-spacecraft communications link, both S-band and X-band equipment was looked at in detail, along with an optical laser system. Other radio bands were ruled out because of general requirement number four, use off-the-shelf hardware were available. For the same reason, the optical laser was ruled out. The laser system would require receiving satellites in Earth orbit [1]. Both an S-band and an X-band system were decided upon. The X-band system will be use as the primary data link. The S-band equipment will be used as a backup system. High frequencies, such as X-band (8.414 GHz) suffer weather effects [2], so the S-band transmitter may be used for very critical data.

Since the data rate capacity of the Deep Space Network (DSN) is about 270 thousand bits per second (kbps) [3], the X-band data rate is set at this value. The S-band data rate is set at 50 kbps. In case of primary transmitter failure, the backup system will still be able to handle

Trade tables

Communications:

	comments
Ka-band	New
X-band	Weather dependant, but off-the-shelf equipment
S-band	Off-the-shelf equipment
Optical laser	Not yet developed

CDC radio frequencies

X-band	(270kbps) Uplink Downlink	 7.161 GHz 8.414 GHz
S-band	(50kbps) Uplink Downlink	 2.1 GHz 2.3 GHz
Mars link (S-band)	(10kbps) Uplink Downlink	 2.1 GHz 2.3 GHz

Mass data storage:

type	data rate (kbps)	access	power/Gbyte (W)	weight/Gbyte (kg)
RAM	8.6	random	6000	34000
Tape drive	250	sequential	80	20
CD RAM	228	random	80	10
Hard disk	3000	random	50	20

ROM store:

type	data rate (kbps)	power/Gbyte (W)	weight/Gbyte (kg)
ROM	150	1300	1300
CD ROM	150	37	9.3

Main memory:

type	comment
Core memory	reliable but heavy and slow
Bubble memory	lighter, faster, uses CMOS control chips
CMOS	newest, lightest, fastest, takes very little power

Architecture:

type	comment
Power CPU	will not be powerful enough for AI software
Bit-slice CPU	very specialized
Parallel CPU's	powerful, fault tolerant, multitasking

Outlined text indicates the trade study choices made.

most of the scientific data collect (with the exception of the SAR data, see the Science subsystem section for more information).

Without a pause to do a trade study, a parabolic high gain antenna was chosen. The problem is in the sizing. To size the communication subsystem, a relation between the antenna's diameter d and it's mass m was derived in the form

$$m=ad^2+b$$

The constants a and b were calculated from two antennas of known size and mass. A similar function was derived for the transmitters, mass versus transmitter power p, in the form

$$m=ap+b$$

where a and b are constants.

A relation between antenna diameter and transmitter power can be found for a given data rate. The equations that define transmitter power as a function of data rate are as follows:

Shannon's law:

$$B=W*\log_2\left(\frac{P_r}{P_n}+1\right)$$

Where B is the data rate in bits per second, W is the bandwidth in hertz, and (P_r/P_n) is the signal to noise ratio (SNR).

$$P_n=kTW$$

Where P_n is the noise power, k is Boltzmann's constant, and T is the background temperature.

$$P_r=P_t\frac{4cD}{(fzd_rd_t\pi)^2}$$

Where Pr is the received power, P_t is the transmitter power, c is the speed of light, D is the distance between transmitter and receive, f is transmission frequency, z is the antenna efficiency, d_r is the receiver antenna diameter, and d_t is the transmitter diameter.

For a given P_t, SNR, and other values, W can be found from:

$$W=\frac{P_r}{(kT*SNR)}$$

These equations can be turned around and used to find P_t for a given data rate B. Note that there is an efficiency factor accompanying the bit rate. Only 40% of the theoretical bit rate can be achieved [4], therefore B is much larger than the data rates quoted above.

When all these equations are used together, a minimum mass configuration can be found. Minimum mass was found for an antenna diameter of 3.66 meters. The validity of this method must be questioned. The mass to diameter and mass to power relations were assumed. Are they really valid? The size of the antenna agrees with the size used by many other spacecraft, such as Voyager (3.6m diameter) [5], and Magellan (3.7m diameter)[6]. The process might be questionable, but the results are in agreement with other spacecraft antenna sizes.

In the interest of cutting mission costs, a 3.66 meter Voyager design antenna will be used.

Trade studies: computers.

When the spacecraft can not transmit data, that data will have to be buffered. The types of mass data storage looked at were hardened RAM, tape, CD RAM (compact laser disk), and hard disk (standard magnetic disk). Hard disks are too sensitive for space borne applications. CD RAM technology is new, and data access times are not yet fast enough for most applications [7]. Hardened RAM is too massive for large memory requirements. Magnetic tape, which has been used extensively in the past, has been chosen. With the new helical scan technology, vary large data capacities are possible, about 35.4 megabits per square inch [8].

The on board AI software and data will need a large amount of random access read only memory. Tape provides sequential memory, and therefore is not a good solution. Hardened ROM is massive. The solution is a CD ROM drive. Unlike CD RAM, CD ROM has been commercially available for awhile. The optical disk is read by a laser beam, so there is no physical contact or degradation of the disk. A CD ROM can hold 540M bytes [9]. Fast data access time also allows programs to run in real time.

The question of software patches has come up. Since even the most extensively checked software and data will have hidden errors, this is a valid question (even in the extensively checked software of Apollo 14, 18 errors were found during it's 10 day mission) [10]. Software patches can still be employed, by use of the tape drives. Write once optical disks may also be considered if the technology proves out.

The main memory of the computers will have to be radiation hardened. There are three basic choices of memory available, old reliable (but slow and heavy) core memory, bubble memory, and CMOS (low power, high speed) memory. CMOS is the best choice.

The main on board computer will be required to run AI software, generally expert systems calling various spacecraft simulations. Incase of an emergency, help is about 44 minutes away, if it is available at all (communications failure). Since mainframes are usually required to run such software in real-time, a very powerful, but light and compact computer will be needed. The computer will also need to be tolerant of faults within itself.

There is a possibility that before the technological deadline, 1998, microprocessors such as the Motorola MC68020 or the Intel 80386 may be available in radiation hardened form. The question is then, are these processors capable of AI on their own? The Mac II, based on the MC68020, is available in a special AI configuration, but it uses a special LISP (a leading AI language) processor chip made by Texas Instruments [11].

Another approach is to build a processor board from hardened components. Specialized machines can be produced with a wide range of power. Memory word sizes can be chosen (usually as a multiple of 4 bits) by using bit-slice ALU's (Arithmetic and Logic Unit) [12].

The last approach explored here is parallel processing (this is the architecture chosen to fulfill the requirements). A number of processors are operated simultaneously to achieve the computing power needed. This configuration is highly fault tolerant, for all the nodes of the processor network are identical. If any fail, the remaining nodes can take up the work, although at a reduced speed. This architecture is sometimes called hypercube (also concurrent) architecture, and can provide the power of a supercomputer in a small box [13]. Computers such as this have been built and tested for aerospace applications. The SANDAC V is a proof-of-principle embedded multiprocessor computer based on the MC68020, and the MC68881 coprocessor (for floating point calculations). The SANDAC V can support up to fifteen processors, each with more than a megabyte of memory. The computer is described as being rugged, or conforming to military ruggedness standards. This does not include radiation hardness, but a radiation hard system could be constructed along similar lines if the MC68020 is not hardened. The SANDAC V also supports software development, by supporting an extensive debugger, the high-level language C, and in the future, possibly ADA. The SANDAC V is rated at 37.5 MIPS (million instructions per second) [14].

Problem areas

One of the largest communication problem areas is accurate antenna pointing. The spacecraft will be on a Martian polar orbit. When the orbital plan is perpendicular to the Mars-Sun line, the antenna will need only slight pointing adjustments. But when the polar orbit is in line with the Mars-Sun line, the antenna will have to be slewed across the sky to maintain pointing.

If people are on Mars, the DSN will not be able to handle the data load. It is estimated that a maximum of about 1,500 kbps would be needed [15]. Alternate communications will be necessary, such as an optical laser Earth-Mars network. Communications would be much more reliable through such a net then by radio to the DSN. The DSN could then be used for other purposes.

Turning to the computer problem areas, verification and validation of the on board AI software will be a major problem. By nature, AI is adaptable, and therefore unpredictable. Even famous expert systems like MYCIN (MYCIN was designed as a tool to aid in the diagnosis and treatment of meningitis and bacteremia infections) have a limited probability of being correct (MYCIN was rated as being 65% correct, compared to human specialists rated from 42.5% to 62.5%) [16].

The on board computers must be hardened against radiation. This is especially so, since the spacecraft will be in a polar orbit of Mars, where the density of charged particles is increased do to the planet's magnetic field.

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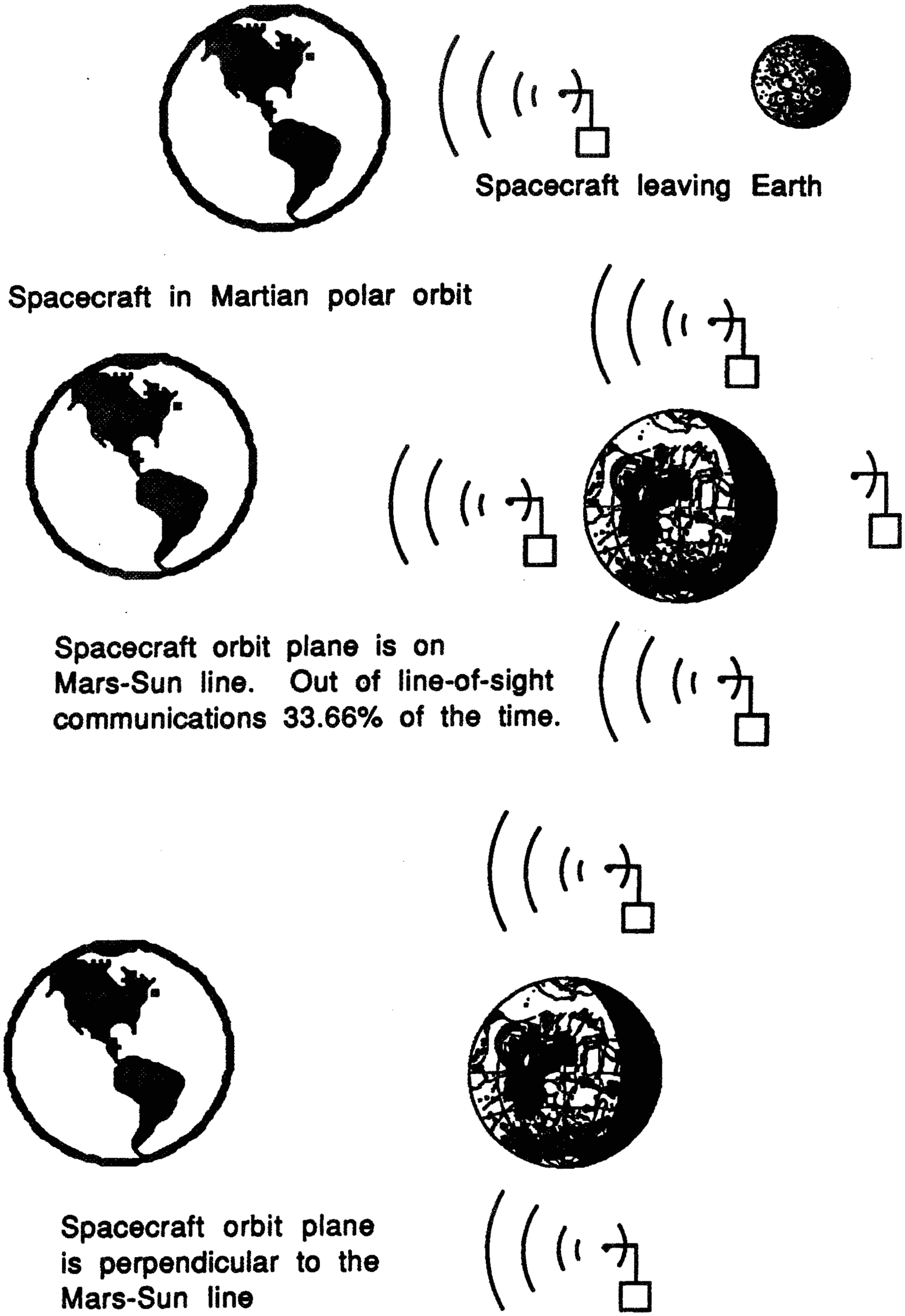

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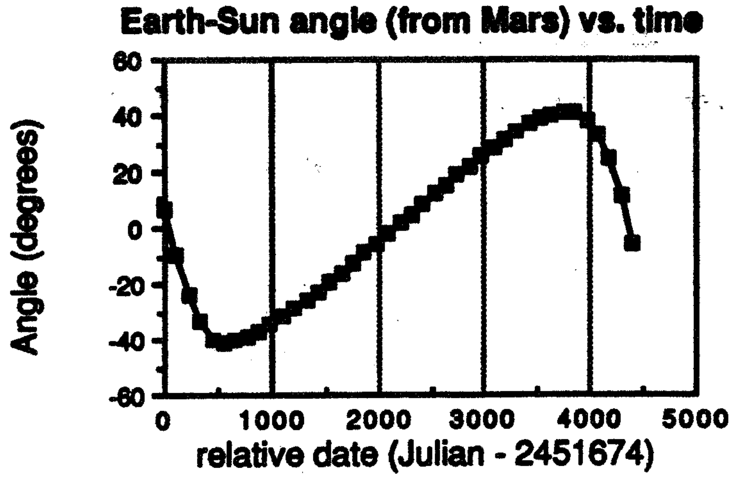

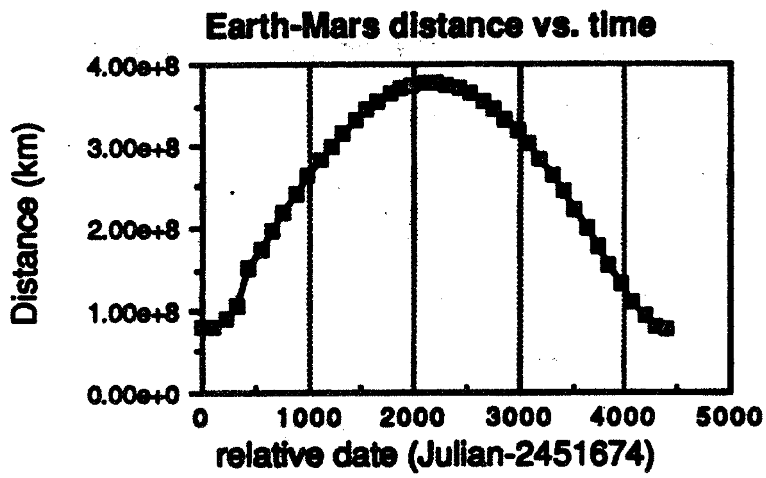

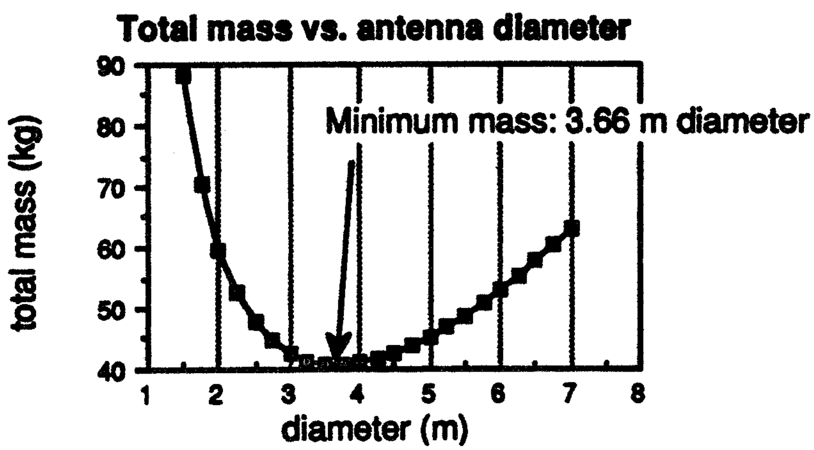

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System overview

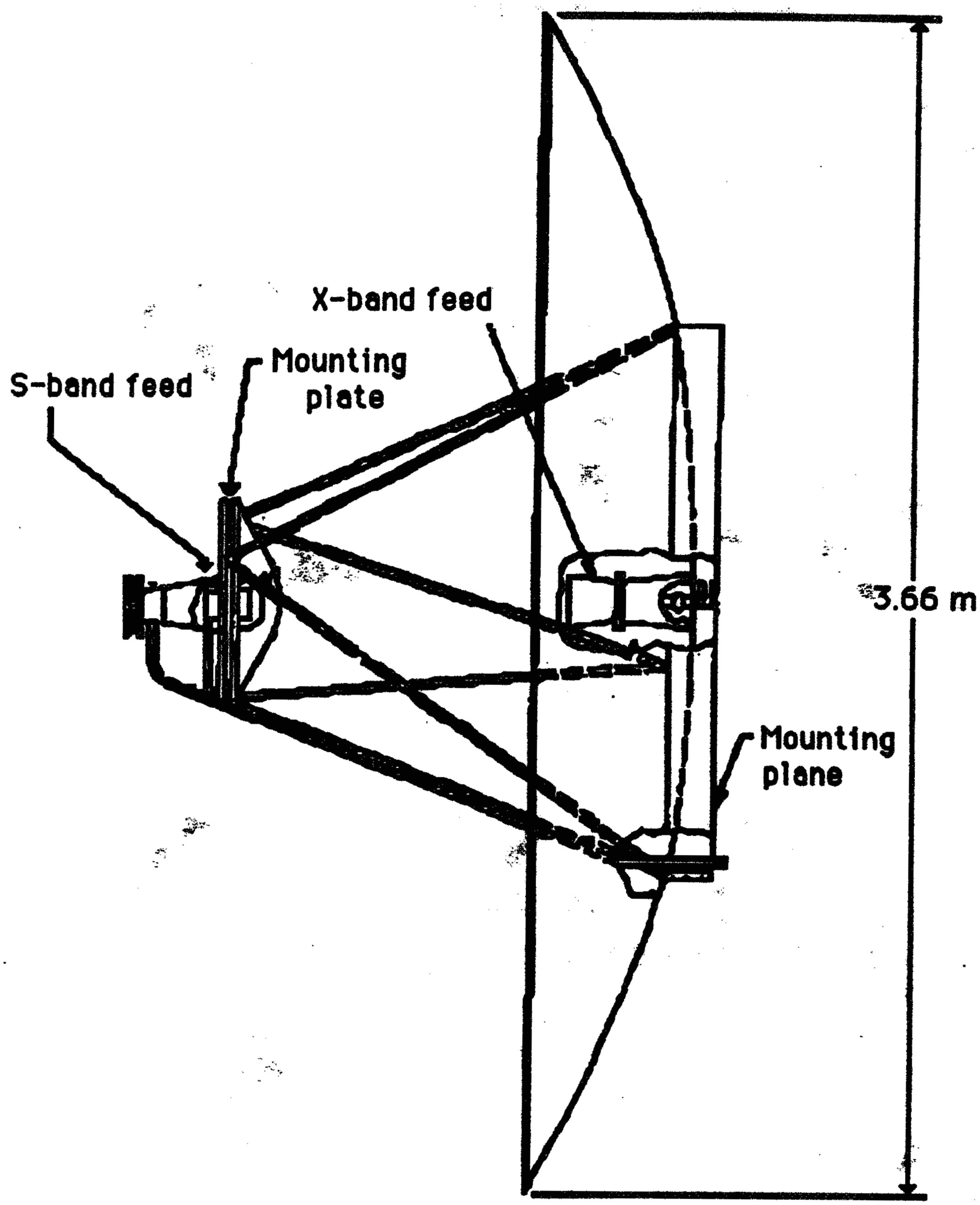
The communication equipment is quite straight forward. There is an X-band primary transmitter system, and an S-band backup system. Both of these systems use the 3.66 meter diameter high gain antenna.

For command uplink, the NASA standard Command Detector Unit (CDU) has been looked at. This CDU standard specifies a maximum uplink data rate of 2 kbps [17]. At this rate, a one megabyte program would take more than 66 minutes to transmit. Since AI programs, and accompanying data, tend to be large, this 2 kbps rate is unsatisfactory. Possibly a new standard will be written.

To comply with specific requirement number three (the spacecraft will act as a relay satellite, and will provide support to the aircraft), a small high gain antenna will be mounted on the spacecraft to communicate with the aircraft. Since needed data rates have not been specified (and the SNR and other critical information about the receiver is not known), the system has not been accurately sized. The antenna will be about 60cm in diameter, and will be used to communicate with the re-entry vehicle during re-entry.

The CDCE is responsible for interfacing all spacecraft components. It will do a partial decode of commands from the ground incase the computer in charge of the system has failed. The CDCE will also be responsible for inter-spacecraft communications. Therefore it will pass data collected from science instrumentation to tape recorders, or to a computer for processing, and then to tape for storage, or directly to the operating transmitter. It will also be responsible for independently spooling buffered data to the transmitter.

The spacecraft secondary computers will be able to provide all basic flight functions, although they will be underpowered to run AI applications. Two such computers will control the re-entry vehicle, both operating simultaneously so that a failure in the primary system can be covered as soon as it is detected. The secondary computers will utilize standard fault tolerant designs to enhance their reliability.



CDC high gain antenna

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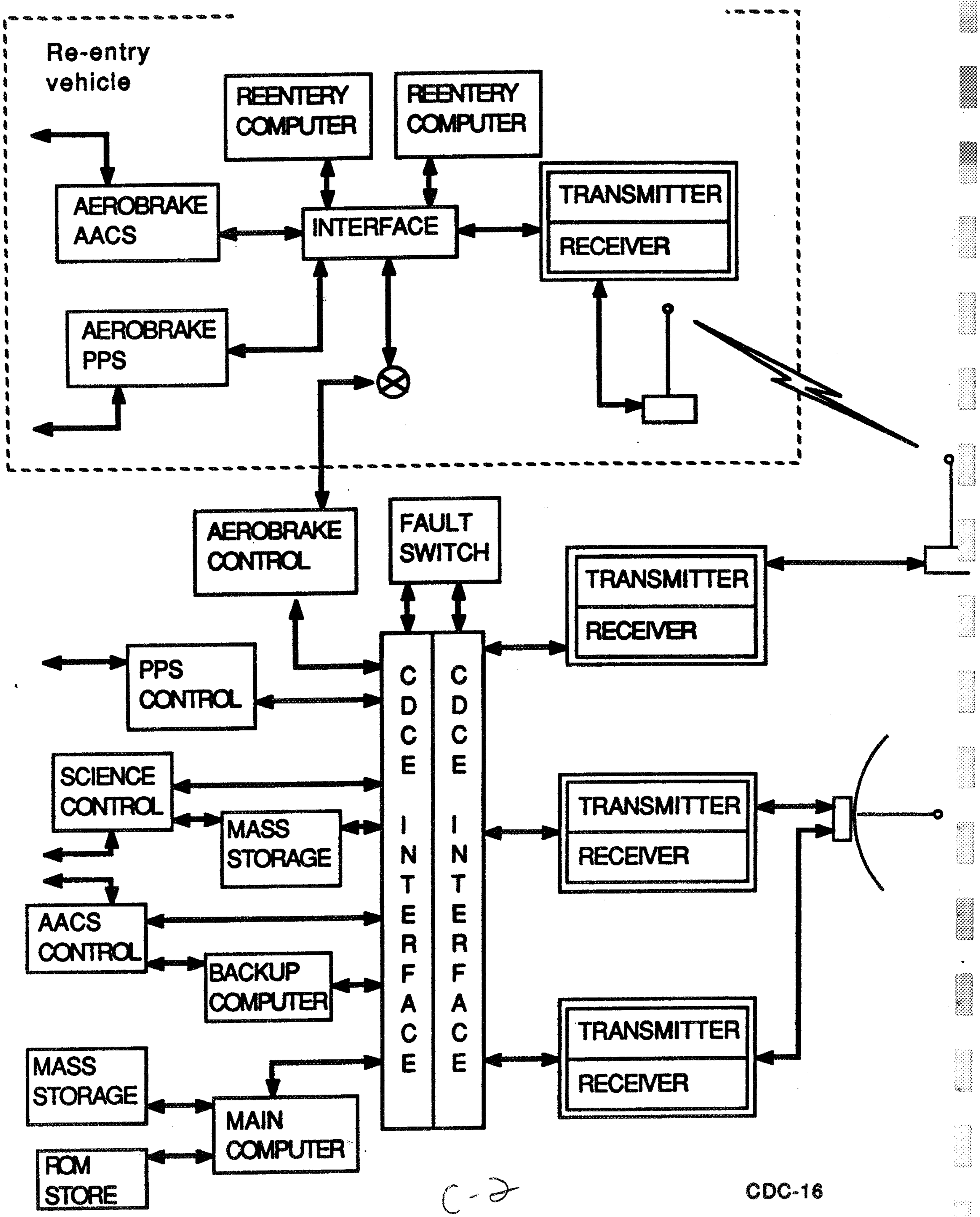
CDC BLOCK DIAGRAM



C-2

CDC mass and power table

Component	#	power	mass	system power	system mass
Mars Comm Link	2				
total		10.0	4.0	20.0	8.0
S-band	1				
transmitter		136.0	13.0		
receiver		14.0	5.0		
total		150.0	18.0	150.0	18.0
X-band	1				
transmitter		59.0	9.0		
receiver		19.0	5.0		
total		74.0	14.0	74.0	14.0
Antenna	1				
total		40.0	19.0	40.0	19.0
Main computer	1				
case		10.0	15.0		
components		200.0	10.0		
CDROM		20.0	5.0		
tape drive		20.0	12.0		
total		250.0	42.0	250.0	42.0
Backup computer	3				
case			3.0		
components		50.0	4.0		
total		50.0	7.0	150.0	21.0
CDCE	2				
tape drive		40.0	18.0		
electronics		5.5	3.5		
total		45.5	21.5	91.0	43.0
TOTAL				775.0	165.0

The primary computer will utilize parallel processing to obtain the speed and power to run AI software. Multitasking will be an integral part of the system, for normal spacecraft functions must be maintained at all times. One ongoing task will be to monitor spacecraft systems, looking for anomalies that may precede an impending failure. The computer will take inputs from various sensors, such as temperature and stress transducers. If faults or failures do occur, the computer will switch to backup systems, or will circumvent the problem in some manner.

The AI software will include a set of knowledge based expert systems, and simulators. If an anomaly arises, a simulator will be used to find possible causes. The AI software will then take appropriate action.

Software will be written for various flight phases, such as: system startup, wait, spiral out, planetary transit, orbit capture, aerobraking, re-entry phase, and science phase.

With the equipment and programs chosen, all the RFP requirements are fulfilled.

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[17] same as [2]

This proposal has satisfied, to the greatest extent possible, all of the mission requirements. If implemented, SPLAT will offer efficient interplanetary transportation for years to come.

FINAL DESIGN REPORT

PROJECT MAVRIC

AAE241 GROUP #2

May 3, 1988

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INTRODUCTION

Manned planet exploration is becoming more prevalent in the space community. Advances in technology are making manned space exploration possible. In the future, man is expected to travel and explore other planets then his own. As expected, the first planet to be explored will probably be Mars. The time frame should be in the earlier 21st century.

The exploration of Mars has already started with the previous fly-by missions and then the Viking Lander. It is reasonable to expect that a manned mission will follow. Initially, man will probably use wheel type vehicles to explore the Martian surface. However the need for further exploration and speed will arise. The logical solution, a manned Mars aircraft.

The goal of this report is to conceptually design a spacecraft capable of safely delivering a Mars airplane from a low-earth orbit to the surface of Mars. The Mars Airplane Vehicle and Reconnaissance Instrument Carrier (MAVRIC) will try to use proven and reasonable technologies to offer the most cost-effective and reliable spacecraft.

Project MAVRIC will utilize an advanced aerobrake system to decelerate into a martian orbit. An instrument bus will then be released to scan the surface for the landing site and the landing conditions. If conditions are unfavorable for landing, the satellite will not commit the reentry system to a final landing. Instead, it will hold the reentry system in orbit until conditions are favorable for a safe landing. Later this satellite will support the aircraft as a communication, scientific, and emergency link with martian ground stations.

PROJECT MAVRIC

In order to start to design a spacecraft to fulfill this mission a Request for Proposal (RFP) was presented. This RFP must be digested to understand the requirements of the spacecraft and specific subsystems. The major requirements include:

1. The spacecraft will consist of two primary components: the payload reentry system and an instrument bus.

2. The following subsystems are defined for system integration purposes:
    a. Aerobrake
    b. Structures
    c. Power and Propulsion
    d. Attitude and Articulation Control
    e. Command and Data Control
    f. Science and Radio Relay Instrumentation
    g. Mission Management, Planning and Costing

3. The spacecraft's components and payload will be delivered to orbit by the Space Shuttle and be assembled on orbit at the space station. The extent of space shuttle support should be minimized.

4. The spacecraft will be able to be retrieved by a remote manipulation device on the space station or the shuttle.

5. Nothing in the spacecraft's design should preclude it from performing several different missions.

6. The spacecraft will have a design lifetime of four years, but nothing in the design should preclude it form exceeding this lifetime.

7. The vehicle will use the latest advances in artificial intelligence where applicable, to enhance mission reliability and costs.

8. The design will stress simplicity, reliability, and low cost.

9. For cost estimating and overall planning, it will be assumed that four space delivery systems will be built.

Three will be flight ready, while the fourth will be retained for use in an integrated ground test system.

10. Mission science objectives are outlined in the document entitled "AAE 241 Mission Science Objectives."

11. The spacecraft should use off-the shelf-hardware where available.

12. The spacecraft should not use materials or techniques expected to be available after 1998.

From these requirements many derived requirements can be found for the specific subsystems. An example would include, the mission planning subtask would have to pick a suitable trajectory to get to Mars. These derived requirements will be discussed in each subsystem's paper.

MISSION PLANNING

The first step for the Mission Management, Planning, and Costing (MMPC) subtask is to distill the RFP to determine all applicable and derived requirements. Then develop a method of attack to begin the design and to also make sure all requirements are satisfied. This method of attack would include distilling RFP requirements, finding derived requirements for each subsystem, research, trade studies, component selection, component integration, and simulation and testing. As shown in Figure 1-1, this method of attack ends up with a completed final design.

Overall, the mission planner is responsible for the total integration of the spacecraft and the aircraft that it will carry. However, some derived requirements for the MMPC subtask include:

1. Selection of a Trajectory

Method of Attack

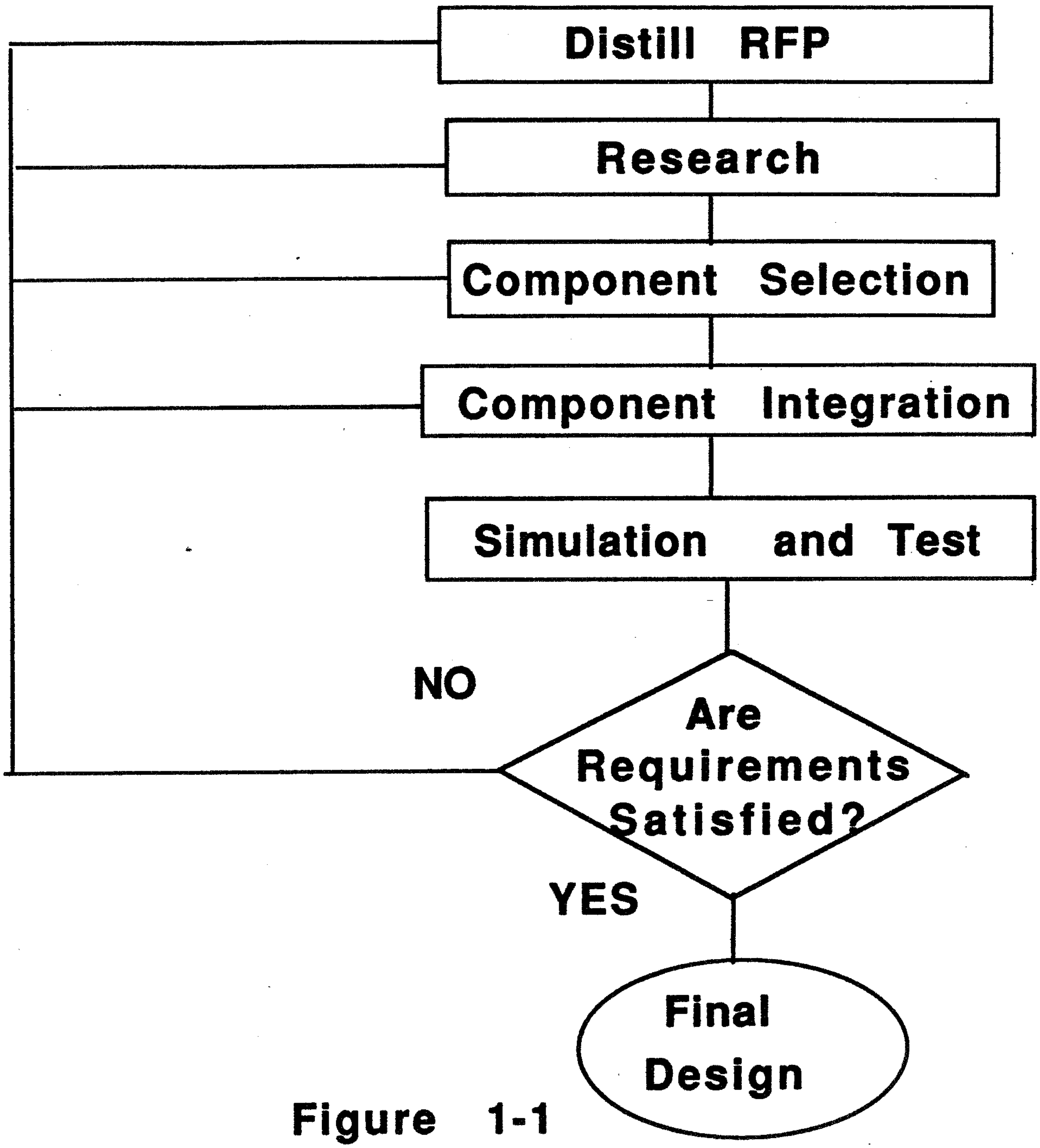

Figure 1-1

2. Calculation of Delta V for the Selected Path
3. Relay Satellite Considerations
4. Mission Timeline
5. Mission Costing
6. Integration with Other Subsystems

SELECTION OF TRAJECTORY

Many different parameters must be considered for trajectory selection. One of the most crucial is the lowest possible delta V required. Others include, thermal considerations, mission time, communications, and availability and flexibility of launch windows. The defined launch window will be May 1, 2003 to August 30, 2005.

Since minimum cost is one of the major design considerations, minimum delta V must be an important consideration in trajectory selection. This is true because even small differences in delta V can result in several thousand kilograms of extra fuel. Thus lta V must be an essential factor in trajectory selection.

Thermal considerations must also be taken into account. It would cost a lot of extra weight to add extensive thermal control systems to protect the spacecraft. This would consequently raise developmental and production costs. Overall, it would be advantageous to pick a path that will not create any serious thermal control problems.

Also mission time must be considered and minimized to an optimum value. First, the shorter the flight time, the shorter the time that expensive ground support technicians and equipment will have to be used. Again, in order to keep costs low, mission time should be kept to a minimum. Furthermore, its always true, the more time spent in flight, the more time that things can go wrong.

Next, communication distances and angles must be minimized. Although the spacecraft will not be doing much on its flight to Mars, it would be much simpler to keep communications with Earth at all times. This would alleviate the problem of storing large amounts of data until it could be relayed to Earth. Also, antenna pointing must be considered. It would be much simpler to have smaller communication angles to help simplify the control system for antenna pointing.

Finally, launch window availability and flexibility must be taken into consideration. This would include when the launch window would occur. For example, the launch date for minimum delta V in the ballistic flight case occurs about every two years. Also, launch window flexibility is an important consideration. As an example of this, would a slip of ten days cause significant increases in delta V? It would be very costly to allow a provision if the launch date slips and large changes delta V occur. In addition, it would be even more costly to wait for the next launch window.

With the type of propulsion systems being considered, the only feasible trajectories are a ballistic or a Venus swing-by mission. Next a trade study is performed to determine the best choice of the two. The results are shown in table form (Figure 1-2).

Specifically, Venus swing-by trajectories yield substantial reductions in delta V for two-way stop over missions. "Conversely, the swing-bys are at best of marginal

TRAJECTORY SELECTION

Trajectories	Venus Swing-By	Ballistic
Delta V using Mulimp	minimum 6.42 KPS	minimum 5.491 KPS
Thermal Considerations	spacecraft gets close to sun, will require active thermal control	not a great problem, insulation
Mission Time using Mulimp	350 days	200 days
Communications	very complex, large communication angles	very simple, small communication angles and distances
Availability of Launch Windows	every 270 days	every two years
Launch Window Flexibility	very inflexible, ten day launch window, great increases in delta V	very flexible, ten day launch window small increases in delta V
Trajectory Chosen		XXXXXXXXXXXXXXXXXXXX XXXXXXXXXXXXXXXXXXXX

Figure 1-2

utility for one-way missions." (Deerwester, 1). As shown in Figure 1-2, one of the major drawbacks of the Venus swing-by is a higher delta V the comparable ballistic trajectories. Also such drawbacks as thermal protection for the spacecraft, communications difficulties, and inflexible launch windows lead to the choice of a straight ballistic trajectory to Mars.

DELTA V CALCULATIONS

After choosing the type of trajectory the delta V and launch date for that trajectory must be calculated. The first step is to define the parking Earth orbit that the spacecraft will be leaving from. Also the orbit that the spacecraft will arrive in at Mars must be determined. After these determinations of an Earth parking orbit of 500 km and a Mars entry orbit of 20,512 km, Mulimp was utilized to find the necessary delta V's. Figure 1-3 illustrates the variation of delta V in the launch window of May 1, 2003 to August 30, 2005. Note that minimum values occur in early June 2003 and late August 2005. The next graph, Figure 1-4, shows an explode view of the daily changes in velocity near the minimum of early June 2003.

Next, the mission time must be determined. Figure 1-5 illustrates delta V vs mission time for a launch in early June 2003. Note that a minimum value occurs approximately at a 180 day mission. After further iterations, a 201.6 day mission time will be utilized.

The final launch date will be June 7, 2003 and the spacecraft will arrive in a martian orbit December 26, 2003

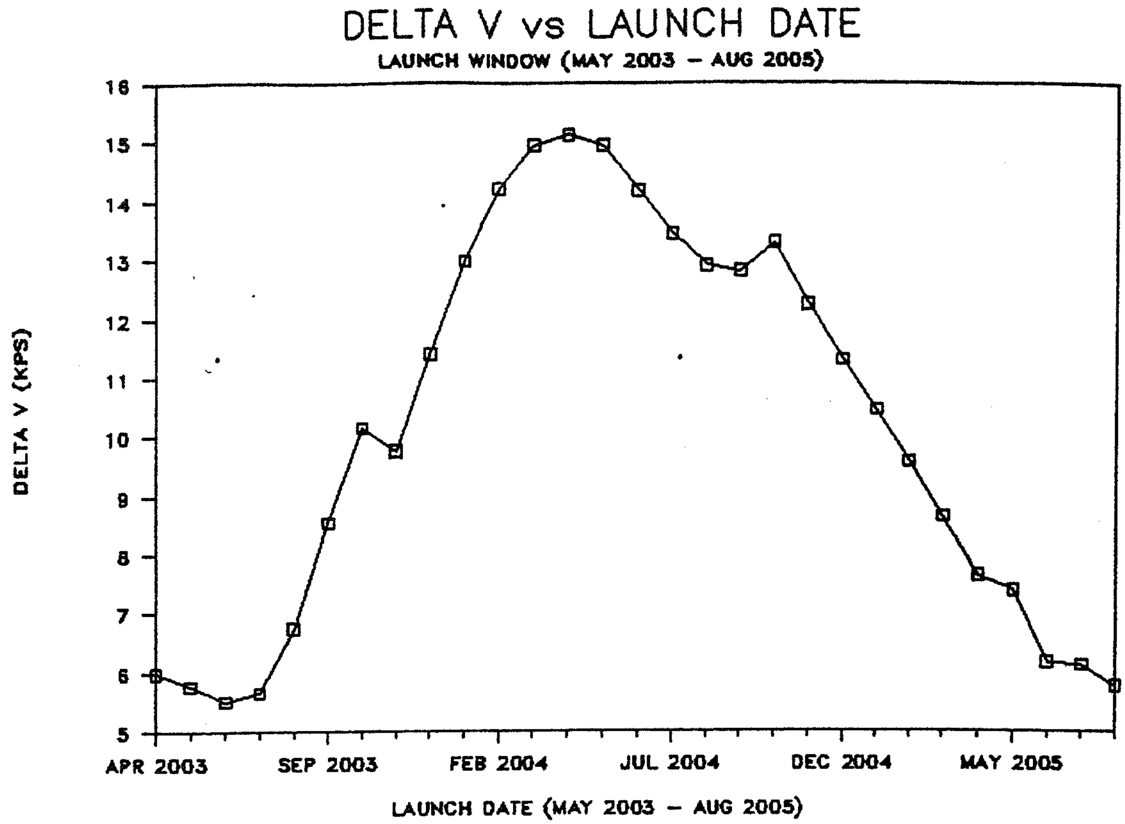

Figure 1-3

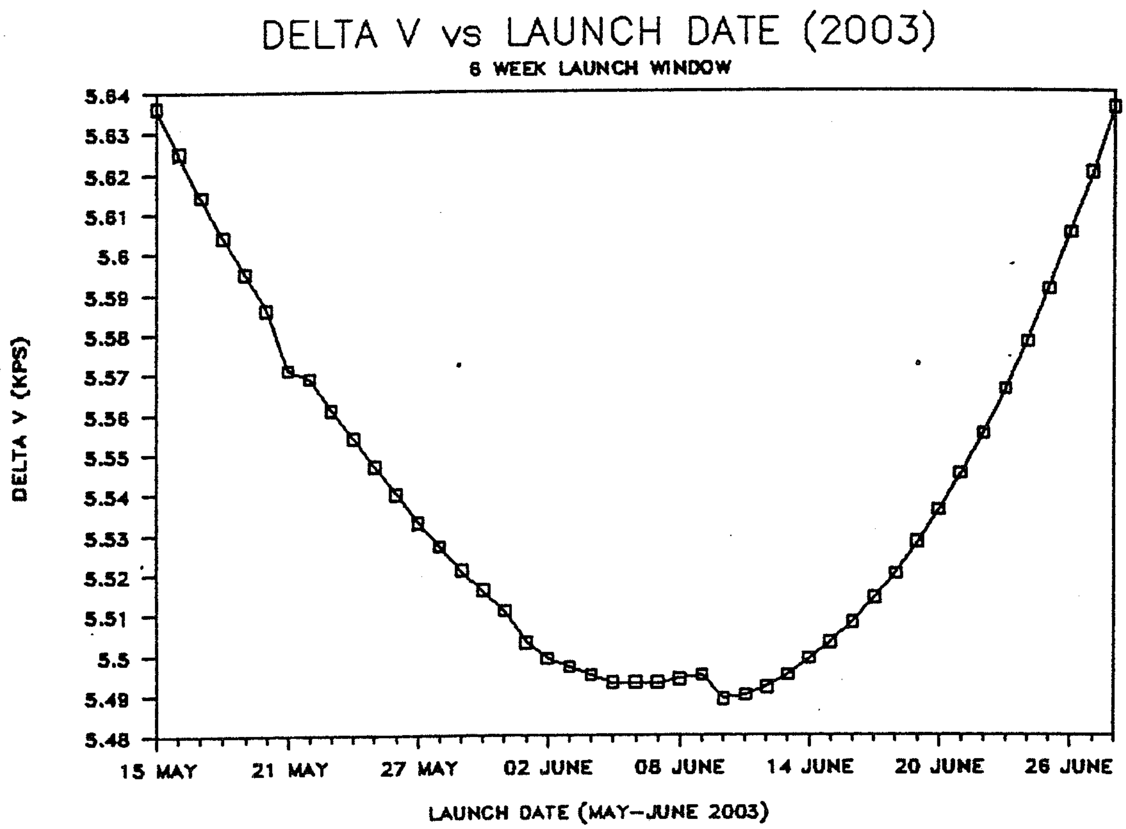

Figure 1-4

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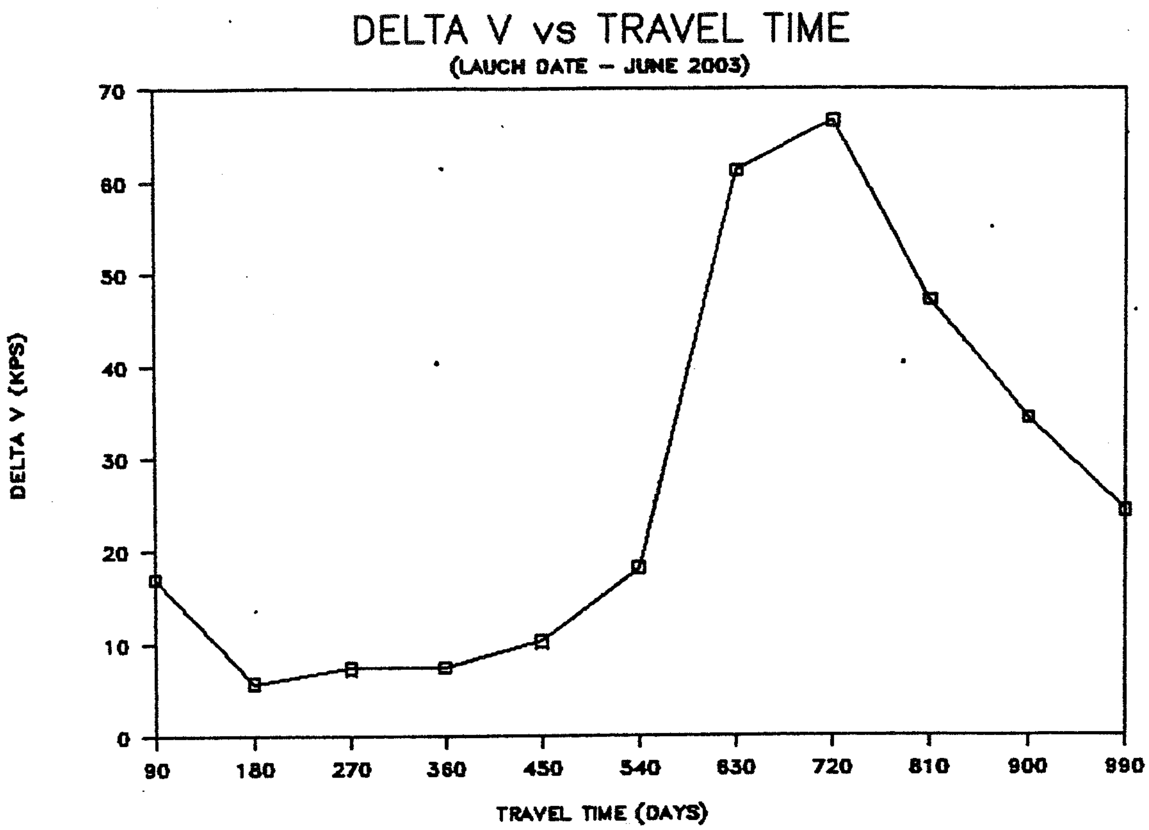

Figure 1-5

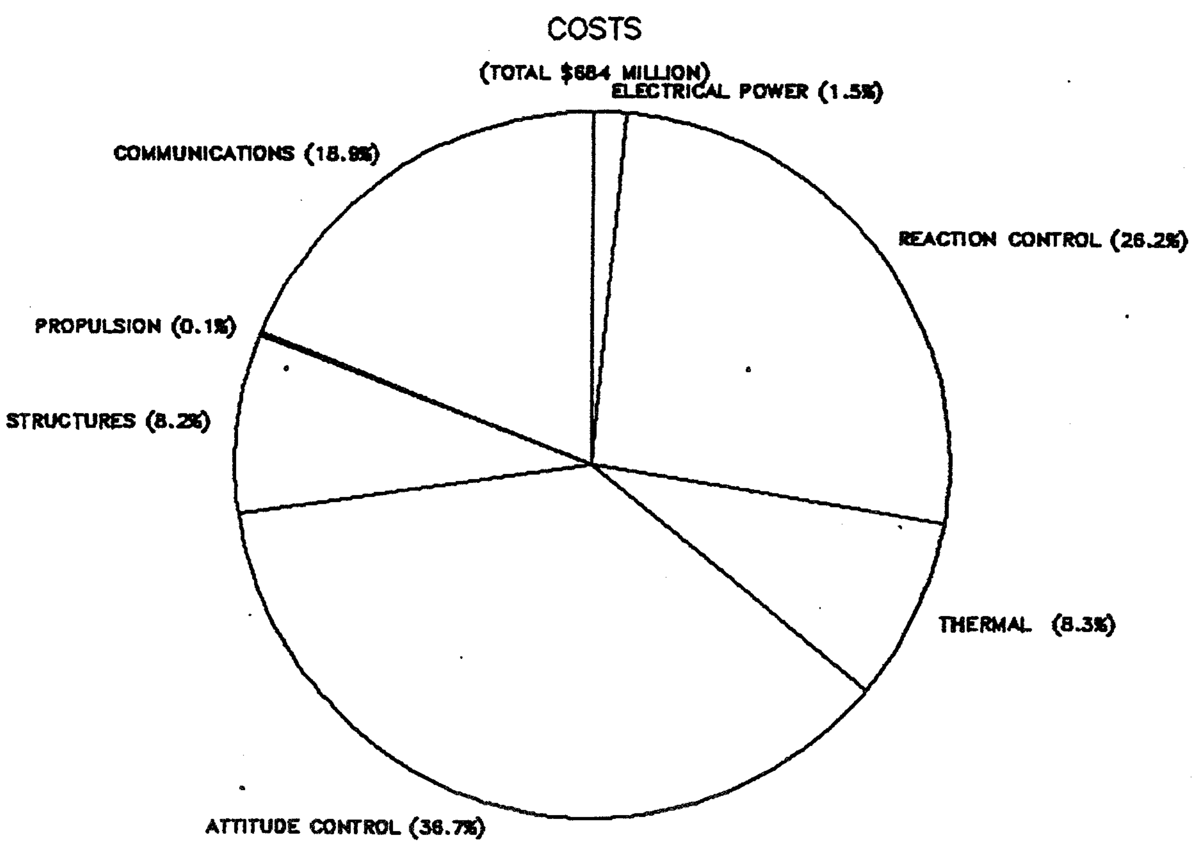

Figure 1-7

with a velocity of 2.5 KPS. The total flight time will be 201.6 days and a delta V of 5.491 kilometers per second (KPS) will be required. The spacecraft will be designed to handle the delta V's within a 10 day launch window of June 7, 2003. This will allow for some flexibility in the launch date with minimum cost and with out any major modifications.

The geometry of the flight path is illustrated in Figure 1-6. Note the communication distance of 1.2 Astronomical Units.

RELAY SATELLITE CONSIDERATIONS

The first consideration for the relay satellite is the selection of orbit around Mars. Because the primary mission of the instrument bus is to support the aircraft's mission, it should be able to communicate with the aircraft at all times. This is because emergency communication will be vital in times of mechanical failure or other mishaps. Thus the question of any orbit besides a synchronous martian orbit is unacceptable to project MAVRIC's design methodology.

Science applications of the instrument bus will have to take second priority to in order support of the aircraft. However, subsequent missions might utilize their instrument bus for a total science application. Since there at least three missions planned, the first instrument bus released could possibly support all of the aircraft delivered to the martian surface, while subsequent relay satellites could support scientific missions.

The way the initial mission is planned, after aerobraking the spacecraft's orbit will be a low circular martian orbit.

MISSION GEOMETRY

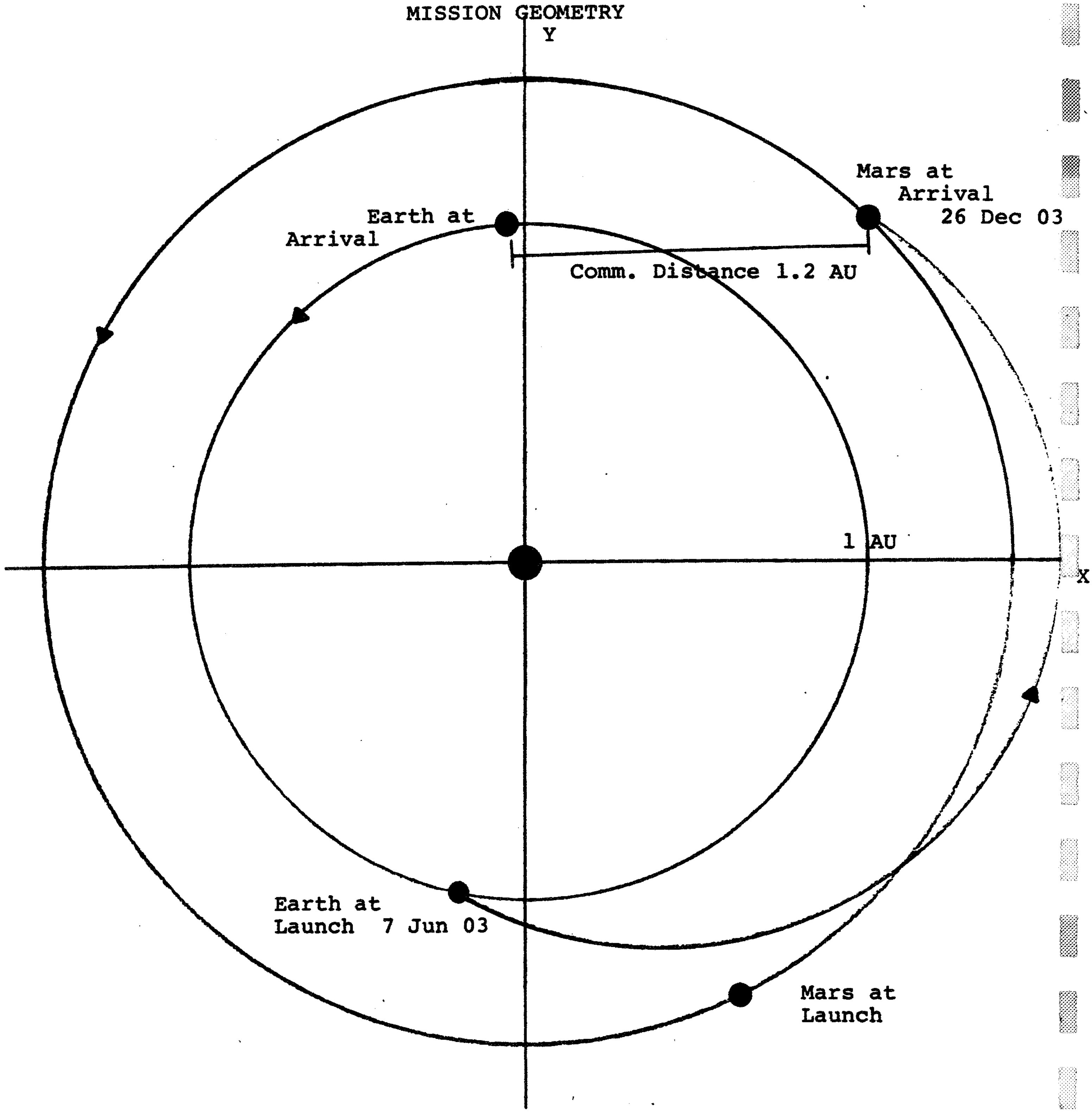

Figure 1-6

The satellite will be released, make a few passes to determine the condition of the landing site, commit the spacecraft to final reentry only if the landing conditions and site are favorable for a safe landing. The instrument bus will then preform a burn to set itself into a synchronous martian orbit. For initial design purposes it was assumed that the landing site would be near the equator of Mars. If this is not the case, extra delta V must be added to make the inclination change to a non-equatorial synchronous orbit.

Appendix 1-A shows all calculations of the altitude of the synchronous orbit (20,512 km). It also goes goes through the calculations for the delta V required to boost the satellite from the low circular orbit to a synchronous orbit. The low circular orbit is 4072 km and a Hohmann transfer was assumed. Also, the eclipse time was needed for both the satellite and the spacecraft in a 4072 km orbit. Assuming that Mars projects a circular shadow (because of its great distance away form the sun) the eclipse time turned out to be 24 minutes. Furthermore the calculation for the satellite's eclipse time in a synchronous martian orbit were also needed. The eclipse time turned out to be approximately 1.22 hours and as in the above cases all work is shown in Appendix 1-A.

MISSION TIMELINE

Then along with the actual mission time, developmental and construction time must be considered. A starting date for initial research should be near 1998. With all research completed and most of the technical problems solved, the initial production of the spacecraft should start in early

2000. The first spacecraft should be ready for shuttle flight in November 2002. This will leave a leeway of two to three months if launching difficulties are encountered with the shuttle. The spacecraft and its payload, the aircraft, will be transported on two shuttle flights to a low earth parking orbit of 500 km. The spacecraft will then require a minimum of assembly and should be ready before the launch date of June 7, 2003.

Right before launch the spacecraft will be towed away from the space station to prevent any damage form the engine exhaust. The engines will then fire and burn for about 34 minutes. After the initial burn the booster engine will be jettisoned and the spacecraft will begin its 201.6 day trip to Mars.

After arriving at Mars on December 26, 2003, it will begin aerobraking. Then, after three days of aerobraking and the final reentry orbit is circularized, the instrument bus will be released. The bus will make passes over the landing site until it confirms that the landing site conditions are favorable for a safe landing. Only then will it commit the spacecraft for final reentry. The satellite will then do a burn and transfer its orbit to a synchronous martian orbit. The spacecraft will then make the final phases of reentry and land safely on Mars. If all goes well, the total mission elapsed time should be 205 days.

MISSION COSTING

Mission costing also must include developmental, hardware, as well as lifetime costs. Using the Lotus program provided by Marshall Space Flight Center, Figure 1-7 was created. Note the total cost of the actual spacecraft will be $684 million and the percentages associated with the listed subsystems. The total lifetime costs for one spacecraft is $1.7 billion dollars making the actual hardware for the spacecraft only one-third of its total life time cost. Assuming four spacecraft were to be built the whole MAVRIC program would cost $6.8 billion. The above figures are for estimation purposes only and may contain some inaccuracies due to the classification of specific MAVRIC components into the categories of communications, electrical power, reaction control, thermal control, attitude control, structures, and propulsion.

INTEGRATION WITH OTHER SUBSYSTEMS

Mission planning subtask must interact and provide information to all the spacecraft's subsystems. This would also include the group providing the payload, the Mars airplane.

The aerobrake subsystem required exact numbers for the entry orbit and velocity around Mars. The entry orbit is also included in the optimization graph for the aerobrake. Thus, it was a two-way interaction in obtaining the optimum value and finalizing it. In addition, the aerobrake subsystem had to provide MMPC with a time for aerobraking (3 days) so that the MMPC could develop an accurate mission timeline.

Thermal control was of great consideration of the structures subsystem. Picking a trajectory that minimized the need for sophisticated thermal control systems was a great concern for structures. It also turned out to be one of the major criteria for trajectory selection because of the weight and costs involved with extensive thermal control systems. MMPC also acted as a liaison between structures and the aircraft section.

Power and propulsion needed exact delta V calculations for the trip to Mars. Eclipse times for the spacecraft during its final circular orbit and an eclipse time for the instrument bus in a synchronous orbit around Mars were also needed. Furthermore, delta V calculations for the transfer orbit of a low circular orbit to a synchronous orbit were also provided.

Requirements for pointing accuracy were collected from the other subtasks and given to the attitude articulation and control (AACS) subsystem. Mission times, trajectories and all details regarding engine burns were provided to the AACS subsystem.

Command Data and Control's (CDC) biggest concerns were communication distances and communications angles. This was also one of the major criteria used in trajectory selection and one of the biggest drawbacks to a Venus swing-by mission. This information helped the CDC subsystem determine antenna size and control system pointing requirements.

The Science and Radio Relay Instrumentation interacted with MMPC on the choosing of an orbit for the instrument bus. Since the mission's objective is to support the aircraft, scientific experiments would have to take a secondary priority. Thus the orbit selection was a synchronous one, centered where the plane would perform most of its operations. This selection required the science subsystem to decide on whether certain experiments could or could not be performed and then choose the specific instruments.

Finally the MMPC acted as a liaison with the aircraft group. Initially they were presented with a maximum packaging size, which they met. They were also given shuttle compatibility data and maximum values for the loads the payload would see during each phase of the mission. The information provided by the aircraft section included, final packaging size, final packaging weight, center of mass of the package, inertia matrix of the package, and material the package was made of. The exact numbers and details will be discussed in the structure subsystem's portion of this paper.

PROBLEM AREAS

One major problem encountered was the details of entering a Martian orbit. It was very hard to tell exactly what plane the spacecraft would enter a martian orbit. It was assumed that it would enter a on the plane of the equator. If this turns out to be different, extra fuel must be added and an additional burn must be preformed for an inclination change.

Also, it was hard to fit the subsystems of MAVRIC into the costing program's categories. Some tradeoffs had to be done to get all of the categories to fit. In addition, even though one of the spacecraft is to serve as a ground test system, costing was done as if all four spacecraft were to be used for space flight. However it does provide a good rough estimate and shouldn't be taken for an accurate figure.

CONCLUSION

All in all, MAVRIC's design has stressed simplicity, reliability, and low cost. MAVRIC utilizes as much off-the-shelf hardware as possible and does not use any technology available before 1998. Rough costing numbers have been included for the whole MAVRIC program of four ships. The conceptual design of MAVRIC stresses the use of present and proven technologies in order to enhance mission reliability and low cost.

APPENDIX 1-A

Synchronous Orbit Calculations

Assume circular orbit

Tmars = 1.03 days = 88992 sec

Mmars = 6.45 x 10^23 kg

G = 6.667 x 10^-11 Nm^2/kg^2

r = (((T^2) * GMmars)/ (4(3.14)^2))^.3333 = 20,512 km

Calculation of Synchronous Orbit Eclipse Time

r = 20,512 km

Rmars = 3393.5 km

tan(theta) = opp/adj = 3393.5/20512

theta = 9.28 deg

eclipse time = (2(theta)/360) * 24.72 hours
= 1.228 hrs

Calculation of Low Circular Orbit Eclipse Time

r = 4072 km

Rmars = 3393.5 km

g = 3.7 m/s^2

V = (g * r)^.5 = 3881.54 m/s

Circum of Orbit = 2(3.14)r = 2.55 x 10^7 m

T = Circum of Orbit / V = 109.85 min

tan(theta) = opp/adj = 3393.5/4072

theta = 39.8 deg

eclipse time = (2(theta)/360) * 109.85 min
= 24.2 min

Calculation of Transfer Delta V

Assume Hohmann Transfer

r1 = 4072 km

r2 = 20512 km

R = r1/r2 = 5.04

mu = 4.305 x 10^4 km^3/s^2

Vc1 = (mu/r1)^.5

Delta V1 = Vc1 ((2R/1+R)^.5 - 1) = .9485 km/s

Delta V2 = Vc1 ((1/R)^.5 - (2/(R+R^2))^.5) = .6146 km/s

Transfer Time = (3.14) * ((r1+r2)^3 / 8mu)^.5
= 5.722 hrs

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AEROBRAKING SUBSYSTEM

JAMES E. WIMPE

I. INTRODUCTION

Upon arrival at Mars on its interplanetary trajectory, the spacecraft must be put in a circular low Martian orbit before deploying its satellite and proceeding to the surface of Mars. To achieve orbit circularization, the spacecraft will use an advanced aerobraking system. This process uses a combination of propulsive burns and atmospheric drag to slow the spacecraft into a circular orbit about Mars.

In a typical aerobraking mission profile, when the spacecraft passes by the planet on its approach hyperbola, a maneuver is done to capture the spacecraft in a highly elliptical orbit about the planet. The periapse of the orbit is then lowered into the upper atmosphere to take advantage of atmospheric drag and gradually circularize the orbit. This phase is controlled by small maneuvers at apoapse to control periapse altitude. Finally, a propulsive burn is applied to circularize the orbit at the desired final altitude.(See Fig. 2-1)

Aerobraking offers considerable advantages over other capture system options: all propulsive, and aerocapture (a single pass through the atmosphere is used to circularize the orbit). It shows mass savings over the all propulsive maneuver and its aeroshell can be reused for reentry. It also is generally easier to control, and is more flexible than the aerocapture system[2.1].

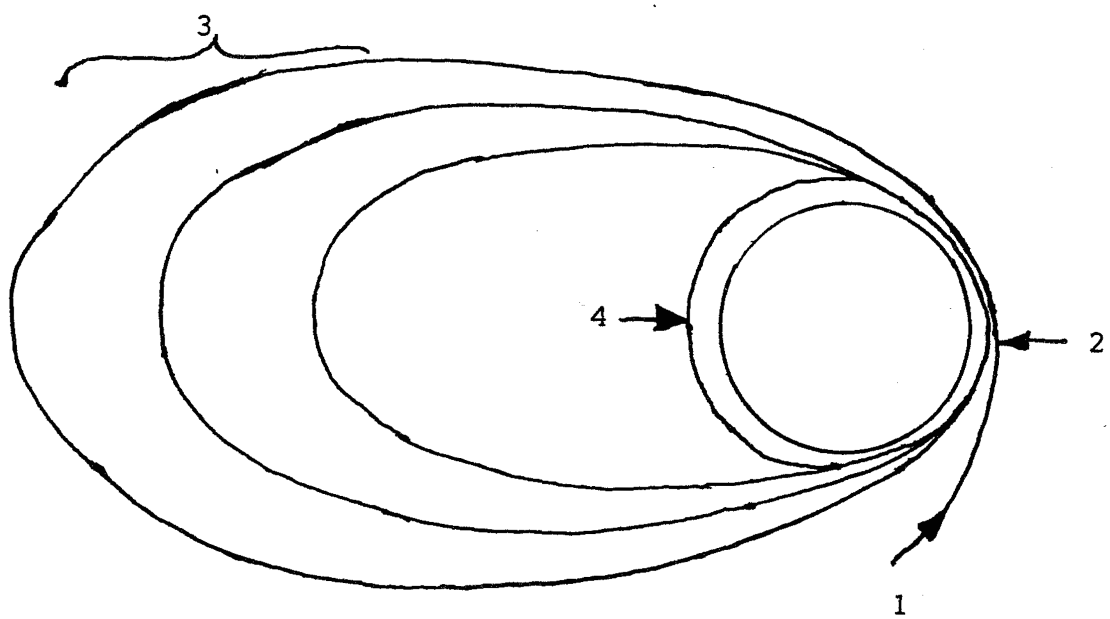

Fig 2-1

1.The approach hyperbola, 2.Capture into elliptical orbit, 3.Control at apoapse, 4.Circularization burn

II. REQUIREMENTS & CONSTRAINTS

There are three major requirements placed on the aerobrake and reentry system. First is the dissipation of orbit energy and circularization of the orbit about Marsas described above. A heat shield and engines will be used for this,as will a temperature sensor and a set of thrusters for control of the system during aerobraking. Second is the protection of the payload and its accompanying satellite during the mission, both during aerobraking, and (for the payload) during reentry and landing. This will also be accomplished by the shield.

Finally, the subsystem must provide for the safe landing of the payload on the surface of Mars. This requirement will be fulfilled through the use of the aerobrake shield as a heat shield during reentry, then the use of a parachute, retro rocket system, and landing gear to achieve a soft landing.

In the design of the aerobrake, several constraints are introduced. They include the maximum time allowed for aerobraking (75 days), a minimum of 3520 Kg must be delivered to Martian orbit after aerobraking (neglecting the mass of the shield). Also, the length of the spacecraft introduces a consrtaint, as illustrated by fig. 2-2. An aerobrake of diameter (D) will effectively protect a conical volume extending 2D from the center of the shield[2.2].

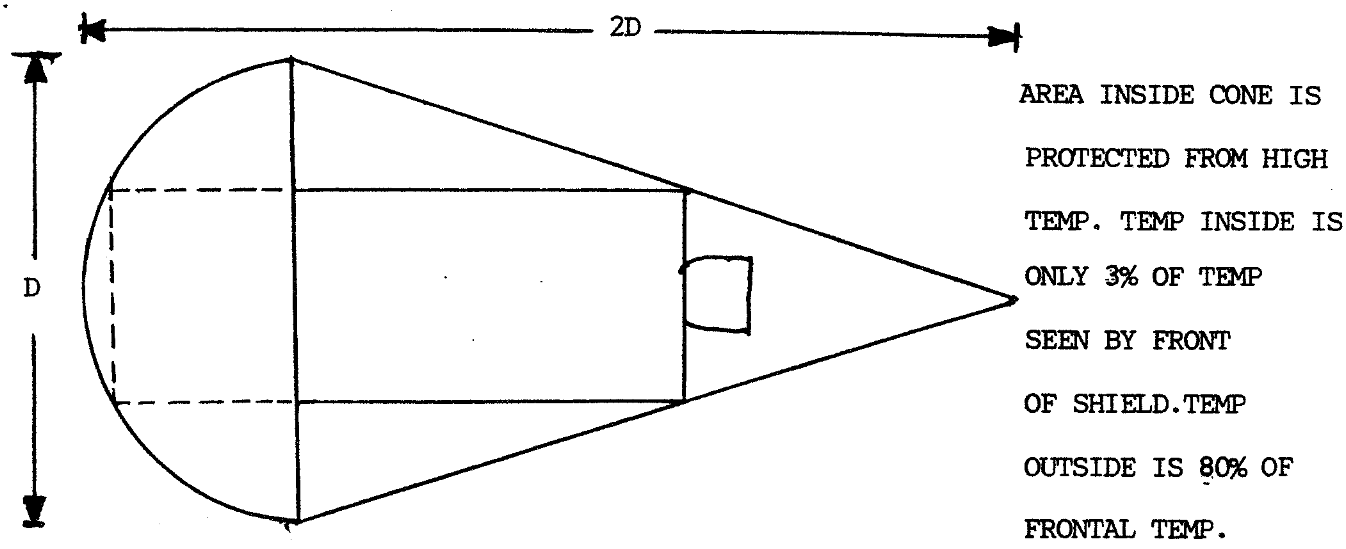

Fig. 2-2

III. AEROBRAKE DESIGN

A. THE AEROBRAKE SHIELD

To aid in determining the required shield size, a computer program called AEROB developed by Dr. Steven Hoffman was used. When various parameters are inputed, the program outputs a shield size and mass, and a final mass in the desired orbit. The parameters to be inputed include the spacecraft mass prior to aerobraking, the specific impulse of the engines to be used, the maximum shield temperature, semi-major axis of the initial elliptical orbit, maximum aerobraking time, and final orbit radius.

The input parameters are shown in fig. 2-3. The spacecraft mass at approach includes: the payload and structure, satellite, engines, tanks, and fuel needed for aerobraking, the reentry system, and the shield itself. The frontal area is the area of the payload and its surrounding structure. The maximum shield temperature is set so that the shield would reach the temperature before impacting on the surface of the planet. The hyperbolic velocity was determined by the trajectory decided upon by mission planning, and the initial periapse was chosen so that the spacecraft entered the the martian atmosphere at that point. The engine parameters follow from the type of propulsion system used. The system uses a bi-propellant propusion (Reference the power and propulsion section of this report for further details of the bi-propellant system). The engine and accompanying tanks are jettisoned after the final maneuver.

INITIAL SPACECRAFT MASS	= 7250 Kg	SPACECRAFT FRONTAL AREA	= 40 M^2
SHIELD COEFFICIENT OF DRAG	= 2.0	AEROBRAKE WEIGHT FACTOR	= 3.85 Kg/M^2
ENGINE I_{sp}	= 320 sec	TANK MASS/ FUEL MASS	= 0.14
ENGINE MASS	= 200 Kg	MAX SHIELD TEMP	= 600·K
SHIELD SPECIFIC HEAT	= 450 J/Kg-K	SHIELD ABSORTIVITY	= 0.7
SHIELD EMMISIVITY	= 0.7	SEMI-MAJOR AXIS	= 35000 Km
TIME LIMIT	= 75 days	WAIT TIME IN INITIAL ORBIT	= 0 days
INITIAL PERIAPSE	= 3400 Km	FINAL ORBIT RADIUS	= 4072 Km
MINIMUM ORBIT PERIOD	= 0.083 days	APPROACH VELOCITY	= 2.5 Km/sec

Fig. 2-3 AEROB INPUT PARAMETERS

A grid search of the program was then done to get an idea of the "terrain" of solutions with the following limits on the grid space:

LOWER LIMIT ON SEMI-MAJOR AXIS	= 45000 Km
UPPER LIMIT ON SEMI-MAJOR AXIS	= 65000 Km
INCREMENT FOR SEMI-MAJOR AXIS	= 10000 Km
LOWER LIMIT FOR SHIELD AREA	= 160 M^2
UPPER LIMIT FOR SHIELD AREA	= 190 M^2
INCREMENT FOR SHIELD AREA	= 5 M^2

The results of this search are shown in fig. 2-4.

Since the total length of the spacecraft at aerobraking is 26 M, the minimum aerobrake aerobrake diameter is 15 M, corresponding to an area of 176.7 M^2. Based on this, a search was then done for shield sizes from 177 M^2 to 179 M^2 using the same semi-major axes. The final mass varies only slightly over this terrain (about 3 Kg), so the area of 178 M^2 was chosen to provide for slightly increased protection over the 177 M^2 size, and it offers mass savings over the 179 M^2 shield. The semi-major axis of 45000 Km was chosen because the time for aerobraking is 3.09 days as compared to 5.18 days for 55000 Km and 6.5 days for 65000 Km. Final aerobraking specifications are given in fig. 2-5.

A preliminary choice for the aerobrake shield material was made based on the maximum temperatures various compounds could withstand. Platinum black coated beryllium oxide with a maximum temperature of 922 K was chosen. This material should be able to withstand aerobraking, where the maximum will be 600 K, and it should also be able to withstand the heat of reentry into the Martian atmosphere, for which the aerobrake shield will be used as the primary heat shield for the spacecraft.

B. CONTROL DURING AEROBRAKING

Control during aerobraking will be achieved through the use of a temperature sensor on the front of the aerobrake shield and the spacecraft's AACS thrusters. The sensor sends data to C^3. If the temperature the sensor reads is over 600 K, the thrusters will fire and the orbit will be raised

Fig. 2-4 AEROB GRID TERRAIN

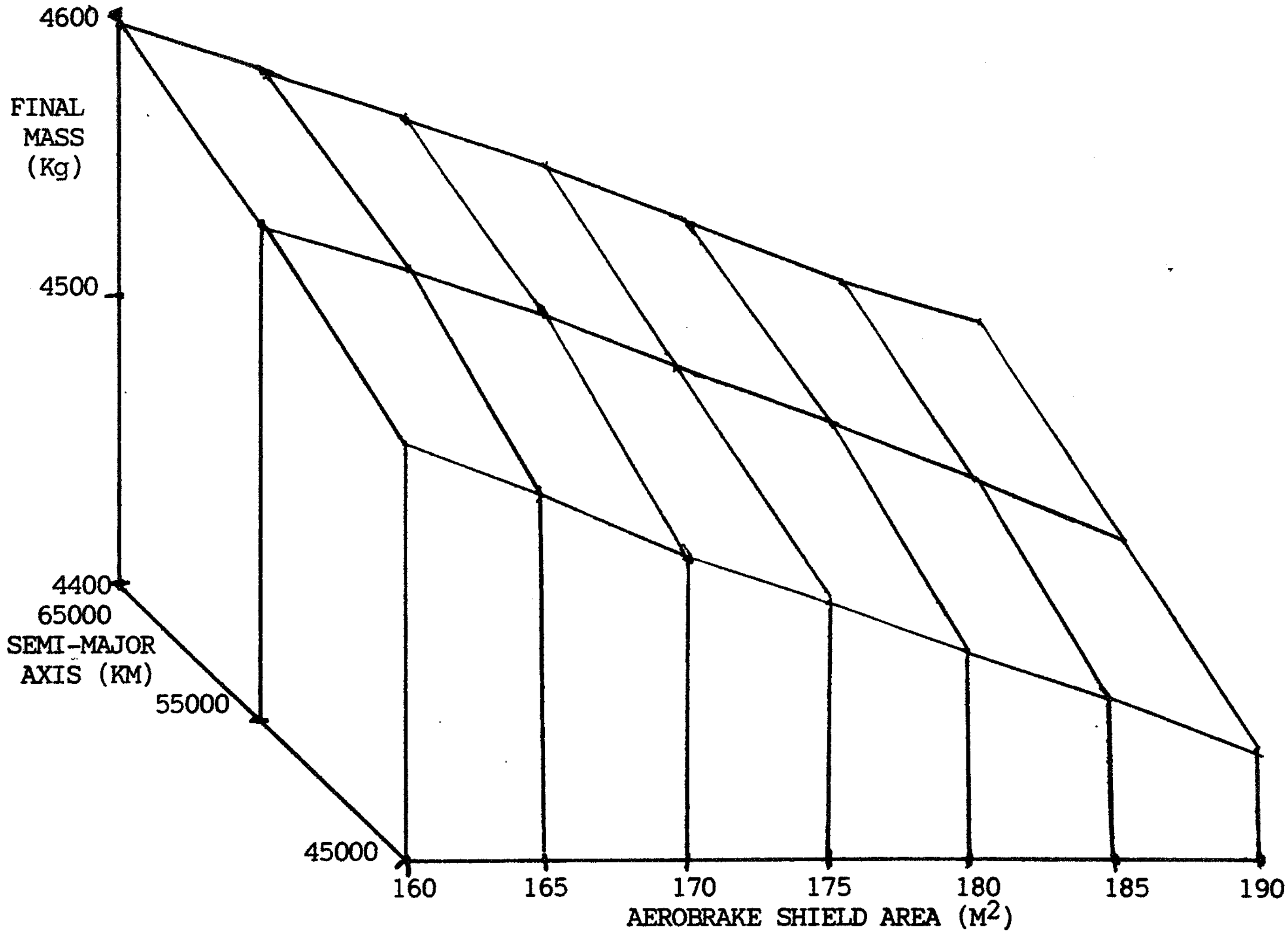

Fig. 2-5 FINAL AEROBRAKE CONFIGURATION

ORBIT	TIME (DAYS)	SCMASS (Kg)	DELTA-M (Kg)	RP (KM)	VP (kM/sec)	DV (M/sec)	A (KM)
0	.00	7250.00		3400.0			45000.0
0	1.67	5664.21	1585.79	3400.0	4.936	774.5	45000.0
1	3.51	5661.98	2.23535	3445.3	4.902	379.52	9496.3
2	3.74	5661.98	.00000	3445.3	4.523	388.89	5449.4
3	3.85	5661.98	.00000	3445.3	4.134	427.65	3825.7

ORBIT CIRCULARIZATION

FINAL DATA

ORBIT	TIME (DAYS)	SCMASS (Kg)	DELTA-M (Kg)	DV (M/sec)
4	3.90	4481.83	258.64	167.5

slightly until the temperature drops below 600 K. Placement of the thrusters is covered in the AACS portion of this report.

C. POWER & PROPULSION REQUIREMENTS

The aerobrake system has only three power requirements during aerobraking. First, power for the temperature is minimal. Second, computer power needs are covered in the power and propulsion section of this report. Finally, thruster power requirements are also covered in the PPS section.

The aerobrake system needs propulsion for three burns: orbit capture, atmosphere insertion on the initial orbit, and final circularization. The propulsion system uses a single engine with an I_{sp} = 320 sec and a mass of 200 Kg. It is a bi-propellant system which uses monometyl hydrazine (MMH) with a nitrogen tetroxide (NTO) oxidizer. AEROB outputs the total amount of fuel needed for each burn (M_f). A table of fuel and tank masses, and tank volumes follows for the aerobraking burns. Reference the PPS section of this report for equations used to reach these figures.

Fig. 2-6 MASSES AND VOLUMES OF FUELS FOR AEROBRAKING BURNS

BURN	M_f (Kg)	M_{NTO}(Kg)	M_{MMH}(Kg)	V_{NTO}(M^3)	V_{MMH}(M^3)	M_{NTO}(tank)	M_{MMH}(tank)
1	1585.79	975.87	609.92	1.1216	0.4262	146.38	91.488
2	2.23535	1.3756	0.8598	0.0016	0.0006	0.2063	0.1290
3	258.64	159.16	99.48	0.1829	0.0695	23.874	14.922
TOT	1846.67	1136.41	710.26	1.3061	0.4963	170.46	106.54

IV. REENTRY-LANDING SYSTEM

Once the spacecraft achieves its circular low-Mars orbit, the satellite is deployed, and the landing site is approved, the landing sequence can begin. The landing system consists of four components:

A. ROCKET FOR DE-ORBIT BURN

B. PARACHUTE SYSTEM

C. LANDING ROCKETS

D. LANDING GEAR

A. ROCKET FOR DE-ORBIT BURN

For the de-orbit burn to begin the reentry sequence, a delta-V of 200 M/sec is to be provided by a bi-propellant engine with an I_{sp} of 340 sec. By using the rocket equation:

$$\text{delta-V} = g_o I_{sp} \ln(M_o/M)$$

where delta-V = 200 M/sec , g_o = 9.81 M/sec^2 , I_{sp} = 340 sec

M = reentry mass = 3520 Kg

M_o, the mass of the spacecraft plus de-orbit rocket and fuel, is found to be: M_o =3738 Kg.

The fuel mass is then: $M_f = M_o - M$

$$M_f = 218 \text{ Kg}$$

Fig. 2-7 FUEL AND TANK MASSES FOR DE-ORBIT ROCKET

	M(Kg)	V(M^3)	M(tank)(Kg)
NTO	134	.1540	18.76
MMH	84	.0586	11.76
TOTAL	218	.2126	30.52

The de-orbit rocket is then jettisoned when its burn is complete.

B. PARACHUTE SYSTEM

Once the de-orbit burn takes place, the spacecraft begins its descent to the surface. Using the equations:

$$D = .5pV^2A^2 \quad \text{-- assuming flat plate drag}$$

$$V_x = V_{ox} - (D/M)\cos B(\text{delta-t}) \ , \ V_y = V_{oy} - (\ g - (D/M)\sin B)(\text{delta-t})$$

where D = drag on spacecraft , p = density of the atmosphere ,

V = velocity of the spacecraft, A = frontal area

V_x = hoizontal veocity , V_y = vertical velocity

M = mass of spacecraft , g = gravity constant

B = $\tan^{-1}(V_y/V_x)$ = angle of spacecraft with the martian surface delta-t = (delta-y)/V_y =time of travel

and an atmospheric profile for the density values[2.3]. At y = 5000 M, the velocity of the spacecraft is found to be:

V_x = 276.05 M/sec , V_y = -186.14 M/sec , V =332.94 M/sec.

Values for the entire descent are in fig. 2-8.

Fig. 2-8 DESCENT DATA FOR Y = 500 Km TO Y = 5000 M

V_o is the vaue of V for a circular orbit of radius R = 4072 Km less the de-orbit burn of 200 M/sec.

V_o = sqrt(Rg) -200 M/sec

where g = 3.7 M/sec^2 , V = 3881 M/sec in the x-direction.

A = 178 M^2 , M = 3520 Kg

y(Km)	p(Kg/M^3)	V_x(M/s)	V_y(M/sec)	V(M/sec)	B(deg)	D/M(M/sec^2)	time (sec)
500	0	3881	0	3881.00	0	0	0
120	9.9×10^{-9}	3880.98	-1678.22	4228.29	23.11	3.411×10^{-3}	453.57
100	8.8×10^{-8}	3880.65	-1722.18	4245.63	23.38	3.045×10^{-2}	465.49
90	$3,815\times10^{-7}$	3879.94	-1743.36	4253.62	23.93	1.331×10^{-1}	471.30
80	1.447×10^{-6}	3876.99	-1763.41	4259.18	24.20	5.067×10^{-1}	477.04
70	5.52×10^{-6}	3866.99	-1779.84	4256.93	24.48	1.938	482.71
60	2.033×10^{-5}	3830.57	-1783.88	4225.58	24.71	7.130	488.33
50	6.995×10^{-5}	3707.65	-1747.39	4098.78	24.97	24.17	493.94
40	2.24×10^{-4}	3330.75	-1590.94	3691.20	25.23	72.84	499.66
30	6.605×10^{-4}	2343.77	-1142.73	2607.50	25.53	174.18	505.94
20	1.806×10^{-3}	474.69	- 263.86	543.09	25.99	237.66	514.69
5	1.104×10^{-2}	276.05	- 186.14	332.94	29.08	25.90	524.75

*DATA FOR y = 490 TO y = 120 IS NOT INCLUDED BECAUSE THE DRAG IS NOT A FACTOR AT THESE ALTITUDES SINCE THE DENSITY IS SO LOW.

To size the parachute, these values at 5000 M were used:

$M = 3520$ Kg, $g=3.7$ M/sec^2 , $B = 29.08$

V = desired final velocity after the chute opens

= 100 M/sec

$p = 1.104 \times 10^{-2}$ Kg/M^3

Along with these equations:

$Mg = D\sin B$ --- by summation of forces

$D = .5pV^2A$ --- again assuming flat plate

The required drag was found to be D = 26644 Newtons.

From that, the area of the chute is $A_{chute} = 427$ M^2.

The system was found to achieve this velocity at y = 1750 M, after a controlled fall by parachute of 26.2 sec.

The parachute is packaged in the rear of the spacecraft and is exposed when the de-orbit rocket is jettisoned. The chute itself is made of Dacron[2.4] and weighs about .67 Kg/M^2 (ref. 2.5) or 286 Kg for the entire parachute. With the aid of a radar altimeter in the shield, the chute is opened by exploding bolts at 5000 M. A small pilot chute opens first, pulling out the main chute which then inflates. At this same moment, the shield is jettisoned so it lands a fair distance from the craft. The chute is then released at at y = 1750 M.

C. LANDING ROCKETS

The landing rockets consist of three bi-propellant engines with $I_{sp} = 320$ sec. They are used to slow the spacecraft to V = 30 M/sec and bring it to a soft landing at that velocity. Using the rocket eqn. in this form:

$$\text{delta-V} = I_{sp} g_1 \ln(M_o/M) - g_2(t-t_o)$$

where $t - t_o$ = 3 sec (using a three-second burn)

delta-V = 70 M/sec

g_1 = 9.81 M/sec^2

g_2 = 3.7 M/sec^2

M_{o1}= spacecraft mass at this point

= 3520 - 685.3 - 286 = 2528.7 Kg

$M_{f1} = M_{01} - M_1$ is found to be M_{f1}= 64.5 kg.

For the constant velocity descent to the ground, delta-V = 0.

using the rocket equation again, with $t - t_o = y_o/V$ = 58.3 sec,

and M_{o2} = 2464.2 Kg, M_{f2} is found to be M_{f2} = 163.6 Kg.

When the spacecraft nears the ground, a final thrust is applied to slow the spacecraft even further. For this maneuver, delta-V = 27 M/sec, $t - t_o$ = 3 sec, and M_{03} = 2301 Kg. Again using the rocket equation, M_{f3} = 27.76 Kg.

Fig. 2-9 FUELS AND TANKAGE DATA FOR THE LANDING ROCKETS

BURN	M_f(Kg)	M_{NTO}(Kg)	M_{MMH}(Kg)	V_{NTO}(M^3)	V_{MMH}(M^3)	M_{NTO}(tank)	M_{MMH}(tank)
1	64.50	39.69	24.81	0.0456	0.0173	5.56	3.47
2	163.60	100.68	62.92	0.1157	0.0440	14.10	8.81
3	27.76	17.08	10.68	0.0196	0.0075	2.39	1.50
TOT	255.86	157.45	98.41	0.1809	0.0688	22.05	13.78

This descent leads to the soft landing of 2273 Kg of spacecraft.

A. LANDING GEAR

At the same time that the spacecraft begins the constant velocity portion of its descent, the landing gear are deployed. They consist of four extendable legs with feet at the bottom (see fig. 2-10). In the case of an uneven landing surface, the legs are extendable to

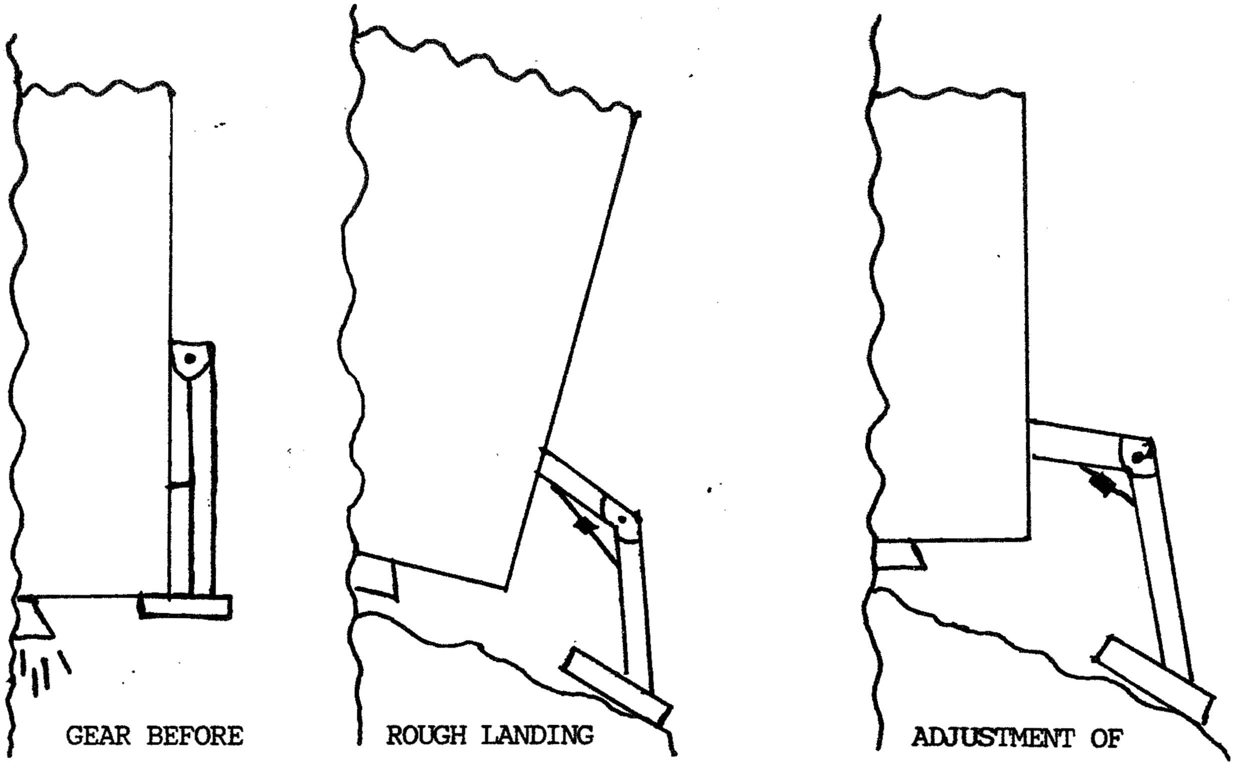

Fig. 2-10 LANDING GEAR CONFIGURATION

return the spacecraft to its upright position. The reentry/landing phase ends with the soft landing of the spacecraft.

V. CONCLUSION

A timeline of the landing sequence from the arrival at Mars to the landing on the surface is given in fig. 2-11. There is also a mass breakdown given for each phase.

This system fulfills each of the requirements placed on it. 1) It dissipates orbit energy by using an advanced aerobrake system. 2) It protects the payload from orbit to landing. 3) It provides for the safe landing of project MAVRIC on the surface of Mars.

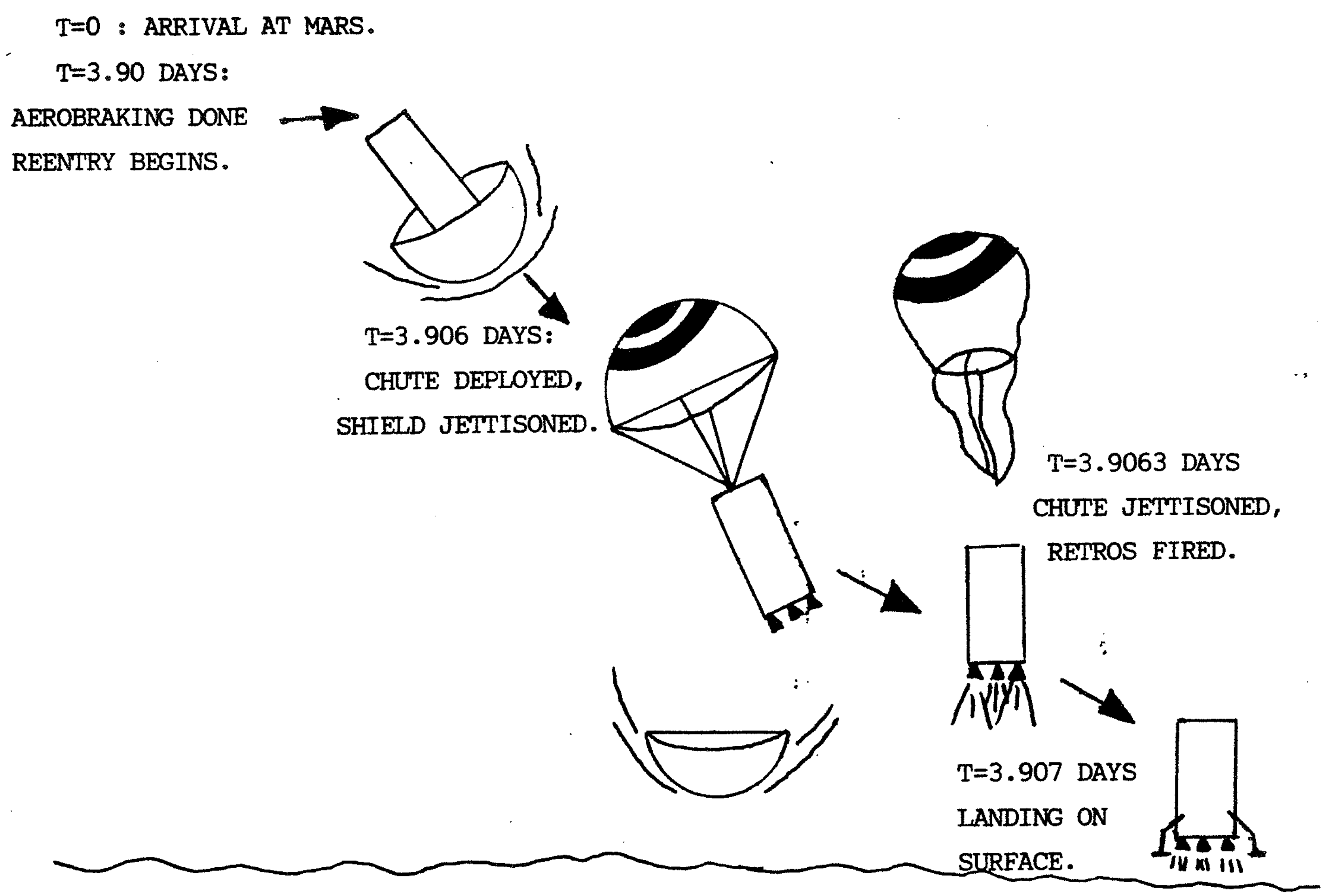

Fig. 2-11 MISSION EVENTS UPON ARRIVAL AT MARS

Fig. 2-12 MASS AT VARIOUS POINTS IN AEROBRAKING/LANDING SEQUENCE

ARRIVAL AT MARS	7250 Kg
AFTER FIRST AEROBRAKE BURN	5664.21 Kg
AFTER SECOND AEROBRAKE BURN	5661.21 Kg
ORBIT CIRCULARIZATION	4481.83 Kg
SATELLITE DEPLOYMENT	3738 Kg
DE-ORBIT BURN COMPLETE	3520 Kg
SHIELD JETTISONED	2834.7 Kg
CHUTE JETTISONED	2548.7 Kg
AFTER FIRST RETRO BURN	2464.2 Kg
AFTER SECOND BURN	2301 Kg
LANDING ON MARS	2273 Kg

VI. REFERENCES

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2.2 Yen, S. M., Aerodynamics and Heating in the Wake of a Shield, Note to AAE 241 Mars Spacecraft Design Group, University of Illinois, March 11,1988.

2.3 "Structure of the Atmosphere of Mars in Summer at Mid-Latitudes", Journal of Geophysical Research.

2.4 Dacron parachute construction from Viking Mars Lander Design.

2.5 de Vries, J. P., and Norton, H. M., "A Mars Sample Return Mission Using A Rover For Sample Aquisition",Case For Mars II, American Astronautical Society, San Diego, 1985.

Structures

Michael Gorden

Introduction

The structures subsystem is responsible for the physical support of all the other subsystems, during every phase of the mission, and for the lifetime of the spacecraft. This particular design is for a Mars transport spacecraft, MAVRIC, which has the specific requirements from the Request for Proposal and Preliminary Design of a Manned Mars Aircraft Space Delivery System. These requirements are as follows.

Requirements

1) Secure all components (utilizing INERT) in locations that:
   - do not affect communications, lines of sight, and pointing requirements
   - position center of mass for application of the Attitude Articulation and Control Subsystem
2) Provide thermal control of material and components

3) Minimize Space Shuttle trips and in space assembly

4) Be capable of with standing applied loads during:
   - Space Shuttle flight
   - interplanetary flight
   - attitude and articulation maneuvers
   - Mars landing procedures

5) Should not use materials or techniques expected to be available after 1998

6) All materials must have a lifetime greater than four years

7) Allow for flexibility in design for future missions with different payloads to different destinations

8) Stress simplicity, reliability, and low cost

Placement of Components

The first step is to construct a base for all the components to be supported by. A simple rectangular truss structure was chosen due to its simplicity, strength, and manufacturing ease. The internal components are to be placed inside of this truss structure. Flow chart 3-1 depicts the design path used in determining the layout of the components. One of the most important design considerations for the spacecraft and the instrument bus is that the components must be secured in locations for application of Attitude and Articulation Control about the center of mass while not affecting communications, line of sight, and pointing requirements.

The internal components are to be placed as close to the center of the truss structure as possible in order for the center of mass to be located inside the truss structure. Placement of the components inside the truss structure are shown in figures 3-2 through 3-7. These components are attached directly to the truss structure. This method allows for the easy movement of components and addition of new components, both of which increase flexibility of the spacecraft to perform various missions. The design also proves to be highly reliable and have a low cost due to the simple fact that not much additional material is needed to make additional compartments.

The spacecraft consists of seven sections. Section one consists of the main engine and its tanks. Section two consists of the engine and fuel tanks for aerobraking maneuvers. Section three consists of the instrument bus and its propulsion system. Section four contains the retro-rockets used during entry into the Mars's atmosphere. Section five contains a parachute, computers, and other various components. Section six consists of the payload. The seventh section consists of the landing rockets and attachment rods for the aerobrake shield. These particular sections are discussed in detail on table 3-1. See fig. 3-1, 3-2, 3-3, 3-4, 3-5, 3-6, and 3-7.

The instrument bus is made in the same basic design as the spacecraft in that it is of a truss design. Components are shown in fig. 3-8. Weights and balances of the components are given on table 3-1.

Boom Deployment

The only boom that needs to be deployed is the magnetometer, all other components will have already been in their operating positions. The magnetometer boom will be of the telescopic type. It will be deployed using springs and explosive bolts. This deployment system was chosen due to its reliability, simplicity, and low cost. Figure 3-8 shows the boom in its deployed state.

Solar Array Deployment

The solar arrays to be used on the spacecraft and instrument bus are Olympus stowed solar arrays. The deployment and packing of these arrays are illustrated in figure 3-9.

Inertia Tensor

The inertia tensor was calculated using the dimensions and masses from table 3-1. The structural weight was determined using an average from past missions. The weight of the thermal control system was determined in the same manner as the structural weight. The components added to the truss structure produced the following inertia tensor, center of mass, and total mass for the spacecraft before the initial delta V burn to begin the flight to Mars.

25288.3062	-206.8325	10.0884
-206.8325	1049823.2629	-144.7907
10.0884	-144.7907	1050185.1126

Total CM (m): X=25.7726 Y=-.0099 Z=.0000

Total Mass (kg): 20,199

The center of mass will move forward as sections of the spacecraft are jettisoned away from the spacecraft. This will produce a more stable spacecraft for the aerobraking maneuvers.

The inertia tensor for the instrument bus was calculated using data given in table 3-1. The components added to the base structure of the instrument bus produce the following inertia tensor, center of mass, and total mass for the structure.

163.7912	-6.7577	10.1290
-6.7577	2094.0881	0.7804
10.1290	0.7804	2121.4877

Total CM (m): X=-0.0473 Y=0.0157 Z=0.0260

Total Mass (kg): 330.58

Shuttle Support

The spacecraft must be contained in the Space Shuttle cargo bay. The requirement to minimize the number of Space Shuttle trips sets up a size and weight constraint in order for our spacecraft to fit into the Shuttle cargo bay, assuming we are only using one trip. The Space Shuttle cargo bay has a diameter of 4.6m and a length of 18.3m with a weight constraint of 22,000 kg.

The antenna, exterior beams for aerobrake shield support, and the aerobrake shield must be broken down in order to fit into the Shuttle cargo bay with the assembled parts of the spacecraft. These components are fastened to the Shuttle cargo bay using trunions and other fastening techniques that are utilized by the Shuttle cargo bay crews. It is stressed that the spacecraft will experience its greatest forces during the Space Shuttle phase of the mission.

	Axial	Lateral
Typical Shuttle Maximum Thrust	3.0 G's	1.0 G's
With Safety Factor, Design Spacecraft Structure for:	6.0 G's	2.0 G's

The spacecraft and the components to be assembled to the spacecraft are positioned into the shuttle cargo bay in such a way that the assembled spacecraft components will be extracted first. The other components will be taken from the shuttle when they are to be attached to the spacecraft. Structural hardpoints will be needed to facilitate the Shuttle's remote manipulator , while it lifts components out of the bay. These hardpoints will be implemented on all components that are to be assembled in space. They will be in positions that best enables the remote manipulator arm to function properly. The components that are to be attached to the spacecraft in the space assembly phase will be attached by as simple of methods as possible. A nut and bolt type of assembly will be used or if future design techniques arise they may also be implemented.

Thermal Control

The spacecraft computers, structural materials, and other various components need to be kept at appropriate temperature levels to function properly throughout the mission. To keep the components at these temperatures a thermal control system had to be chosen. During the choosing process simplicity and low cost were stressed. Also a thermal control system must be protected and stable in all environments encountered during the mission.

Thermal control can be accomplished by using a passive of active system. Passive control is accomplished by using: thermal coatings with low solar absorbance and high emittence, thermal insulation, heat sinks, phase change materials, and excess heat that is produced by various components of the spacecraft. Active control is accomplish using heat pipes, louvers, and heaters. A trade study was conducted in order to find the best type or combination of thermal control systems to be used.

	Passive	Active
Advantages	zero power input low cost high reliability simplicity light weight	precise temperature control
Disadvantages	less control of temperature range	high cost complex control

A passive control system was concluded to be the best since it adhered to the established requirements. Specifically it is cost efficient and simple, but it is still able to keep the spacecraft within an acceptable temperature range. The only exception to this is all passive system will be the instrument bus, in which a passive and active combination will be used.

The thermal control system for the spacecraft will utilize thermal blankets and thermal insulation. Both will use Aluminized Kapton since it has an absorbtivity of 0.35 and an emittence of 0.6. The thermal insulation will be covered by Aluminum plates and thermal blankets. These Aluminum plates will help in heating the interior of the spacecraft because the conduct heat readily. Exposed appendages will be covered by Aluminum tape since it has a small absorbtivity of 0.12. White paints maybe used on the spacecraft in areas that are exposed to the sun for extended periods of time.

The interior of the spacecraft will be kept at an appropriate temperature by varying the thickness of the thermal insulation and thermal blankets. Certain components in the spacecraft will also be packed in insulation for their protection. Since no heat will be produced by the electrical components during the interplanetary phase of the mission (control is accomplished by the instrument bus computer), no active techniques are needed to radiate heat from or to the interior of the spacecraft.

The instrument bus will be using a combination of passive and thermal control devices. It will be using the same types of passive control as those used on the spacecraft. Since there is interior heating due to the electrical components on the bus a tighter range of temperature is needed. One of the most reliable and simple active control system is the louver system. The venetian blind type of louvers were chosen because of their simplicity. They are opened and closed by bimettallic thermal springs that expand and contract at given temperatures. When they are open, a material that has a low absorbtivity and high emittence is exposed. When they are closed a material with both low absorbtivity and emittence is exposed to the sun. This particular combination enables the bus to be kept at a small range of temperatures during all of phase of its mission.

Material Selection

The decision on the material to be use for structural support was obtained from trade studies of numerous materials that are used for structural applications. A trade study was conducted on five materials that are used in the aerospace world. This particular trade study is shown in figure 3-10. The material selected for the truss structure and the aerobrake support structure is Aluminum 7075. This material was selected because of the following reasons: 1)low cost, 2)good fatigue life, 3)easily fabricated, 4)good strength to weight ratio.

A composite material is desired for the booms due to its high strength to weight ratio. Although they are usually high in cost, composites do have a weight savings and therefore a cost savings can be obtained. It is stressed that only a small amount of composites are used on the spacecraft.

Interaction During the Mission

During the Mars spacecraft mission certain sections of the spacecraft will be jettisoned. This will be done by exploding bolts at key locations on the spacecraft. Care must be taken to make sure the discarded sections do not damage the existing spacecraft during their deployment.

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3) Kirkpatrick, J.P. and Brennam, P.J., "Advanced Thermal Control Flight Experiments," Thermophysics and Spacecraft Thermal Control, pp. 409-429, 1979.

4) Vajta, T.F., "Thermal Control Materials," Space Materials Handbook, pp. 95-177, 1965.

TABLE 3-1 Weights and Dimensions

Component	Mass (kg)	Dimensions (m)
Section 1		
1. engine & tanks	1650	# 2.5x1.0x1.5
2. LOX fuel	9429	r = 1.254
3. LH2 fuel	1571	* r = 1.38
Section 2		
1. engine & tanks	225	# 1.0x1.0x1.0
2. MMH fuel	577	* r = .338
3. NTO fuel	923	r = .54
Section 3		
1. instrument bus	330	2.0x2.0x2.0
2. engine & tanks	108	# 0.5x0.5x0.5
3. MMH fuel	269.2	* r = .42
4. NTO fuel	430.8	r = .416
5. He pressurizer	28.25	* r = .363
Section 4		
1. engine and tanks	35	# 0.5x0.5x0.5
2. MMH fuel	90.4	* r = .292
3. NTO fuel	144.6	* r = .289
4. He pressurizer	9.5	* r = .252
Section 5		
1. parachute	300	1.0x1.0x1.0
2. solar arrays	5	* 1.1x1.0x.03
3. fuel cell	13	0.5x0.2x0.2
4. computer system	15	0.5x0.5x0.5
5. gyros pack	16	0.3x0.3x0.3
6. power control unit	15	.25x.25x.25
7. AAC thruster tanks	12.5	--
8. MMH fuel	30.8	r = 0.2
9. NTO fuel	49.2	r = 0.2
10. He pressurizer	3.3	r = 0.18
Section 6		
1. payload	1453	D=4.5 L=18
2. star mappers	7.2	* r = .178
3. AAC thrusters	16	* .05x.04x.04
4. S-band antenna	18	area = 4m2
Section 7		
1. aerobrake shield	689	r = 7.55
2. landing rockets & tanks	44.3	* 0.5x0.5x0.5
3. MMH fuel	98.4	r = 0.3
4. NTO fuel	157.4	r = 0.3
5. He pressurizer	10.5	r = 0.26
6. landing structure	50	

TOTAL DRY WEIGHT = 5000kg + 100kg(contengency mass) = 5100 kg
Structure Weight (20% Dry Weight) = 1020 kg
Thermal Control (5% Dry Weight) = 255 kg
TOTAL SPACECRAFT WEIGHT = 20,199 kg

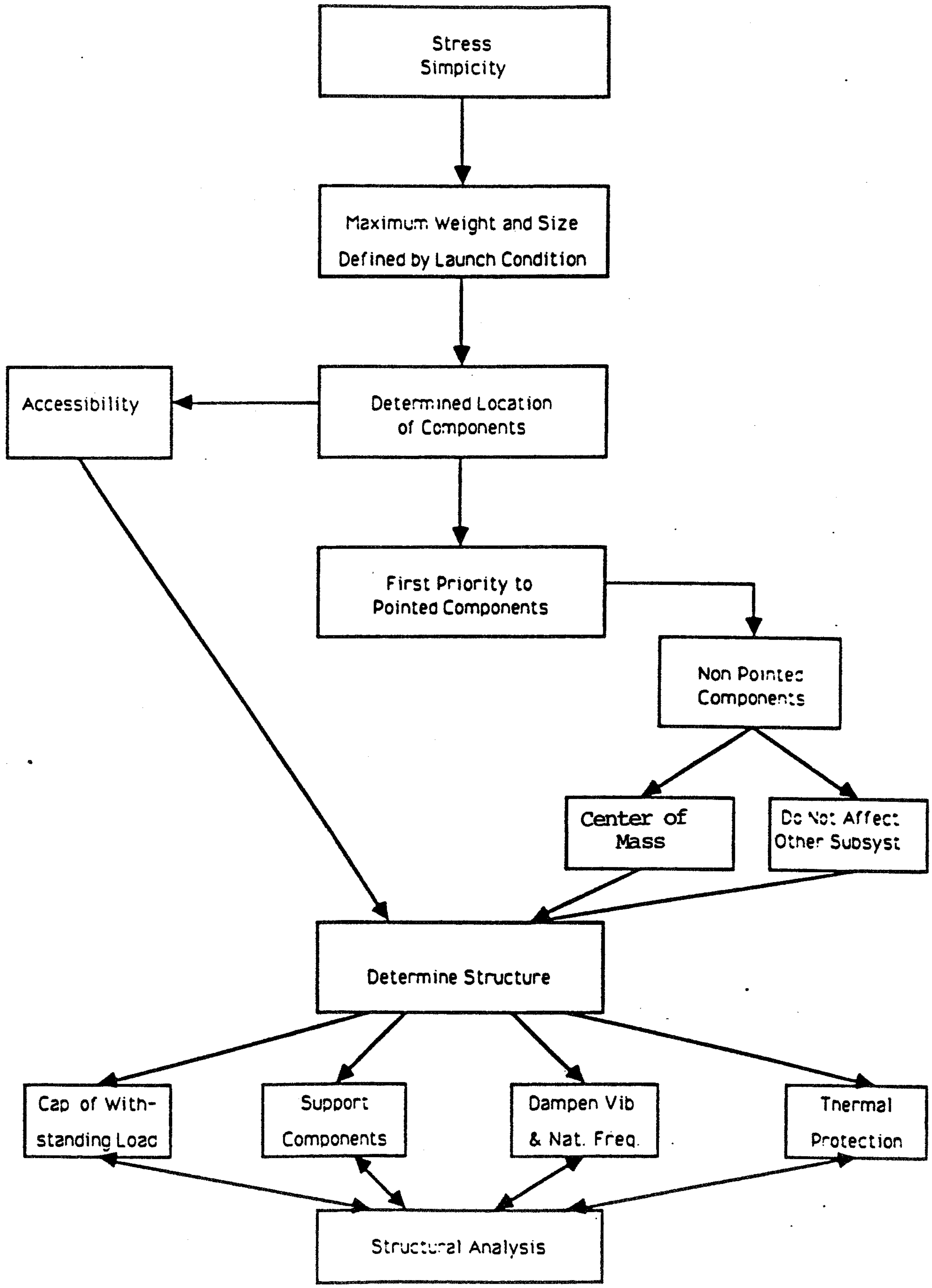
Instrument Bus

1.	gyro pack	16	0.3x0.3x0.3
2.	reaction wheels	6.8	D=.35 L=.15
3.	sun sensors	1.3	* .11x.12x.07
4.	horizon sensors	1.3	* .11x.12x.07
5.	computer system	25	0.5x0.5x0.5
6.	science instruments	97	--
7.	solar arrays	15	* 2x2.15x.03
8.	fuel cell	10	0.5x.15x0.2
9.	power control unit	30	0.4x0.4x0.4
10.	AAC thrusters	31	* .08x.04x.04
11.	AAC tanks	5.0	--
12.	MMH fuel	5.4	r = .11
13.	NTO fuel	8.4	r = .11
14.	He pressurizer	.55	r = .10
15.	X-band antenna	1.0	D=.02 L=.25
16.	thermal control	15.8	--
17.	structure	61	--
	TOTAL	330.58	

- dimensions are of engine only
* - more than one component with same dimensions
(refer to figures)

DESIGN of STRUCTURES

Flow Chart 3-1



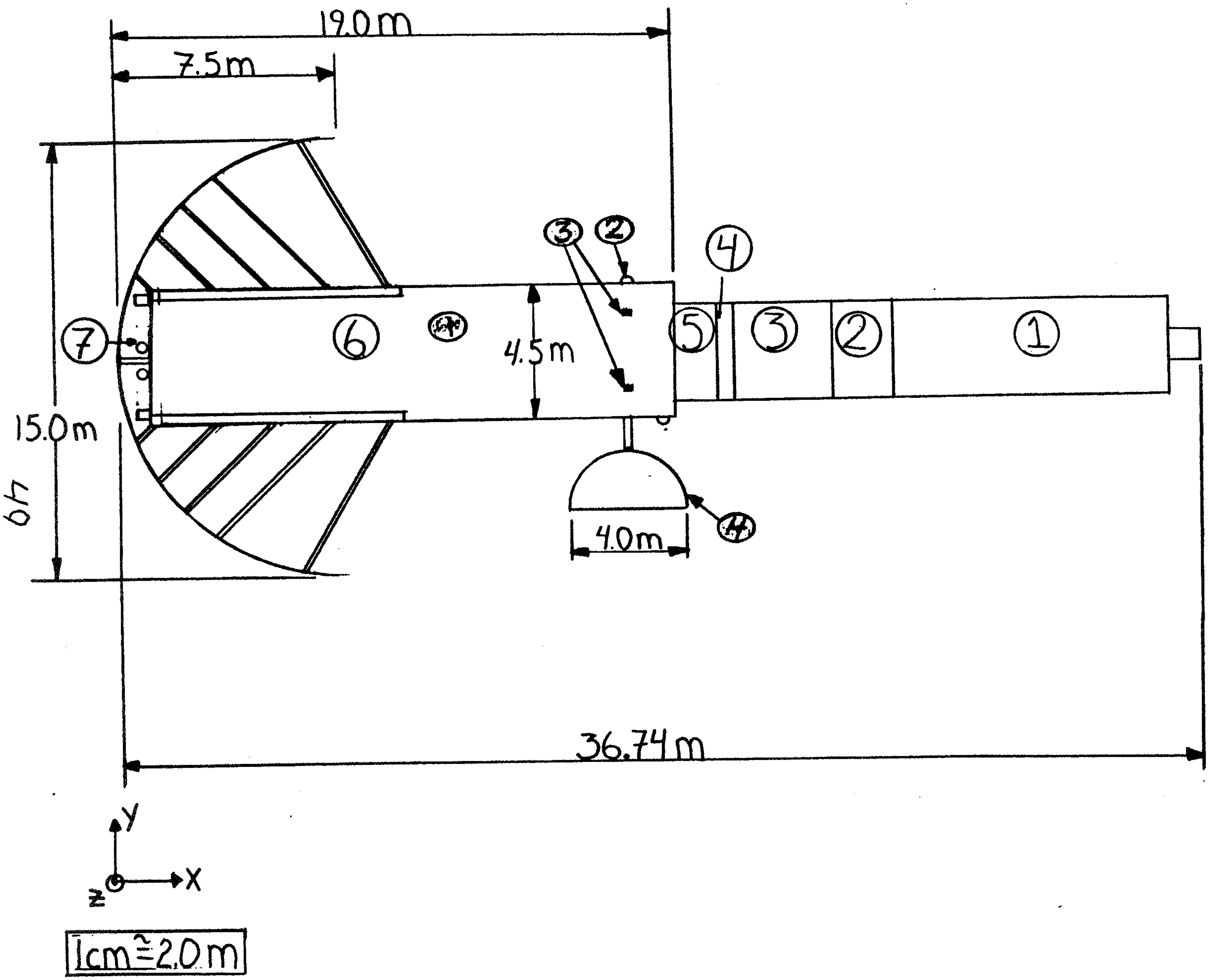
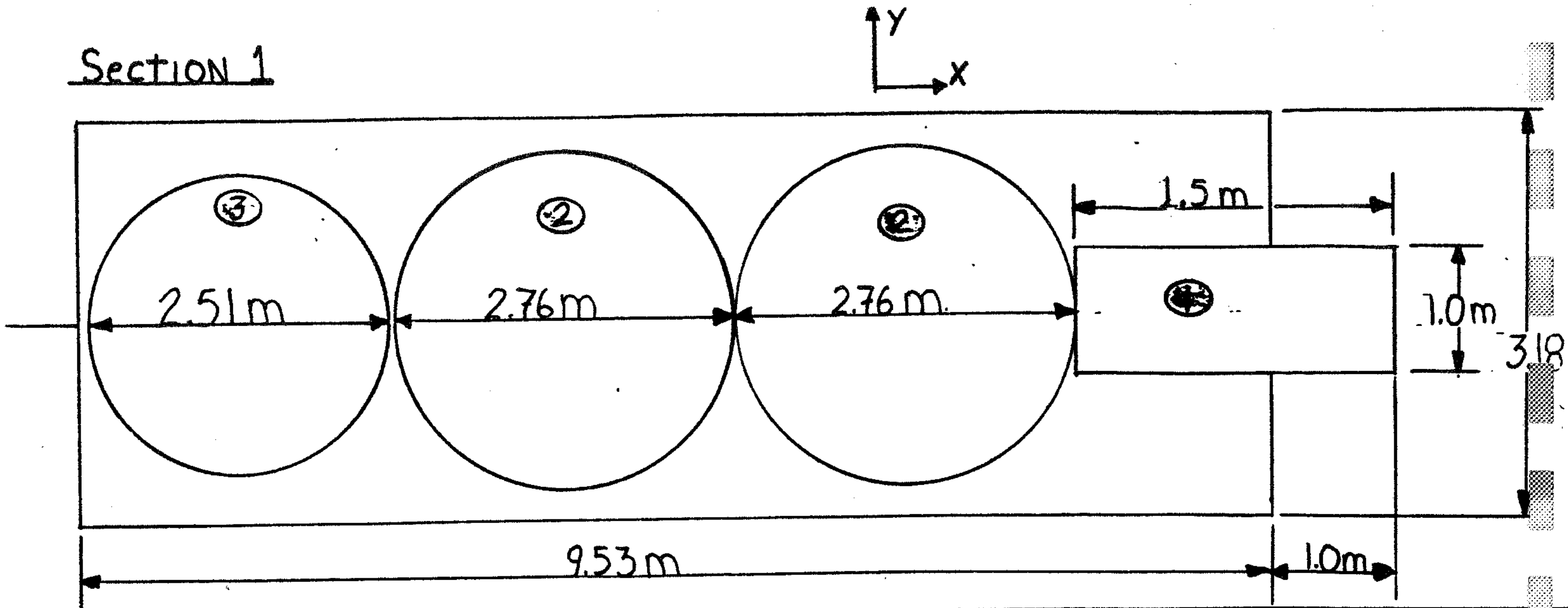


Figure 3-1

refer to Table 3-



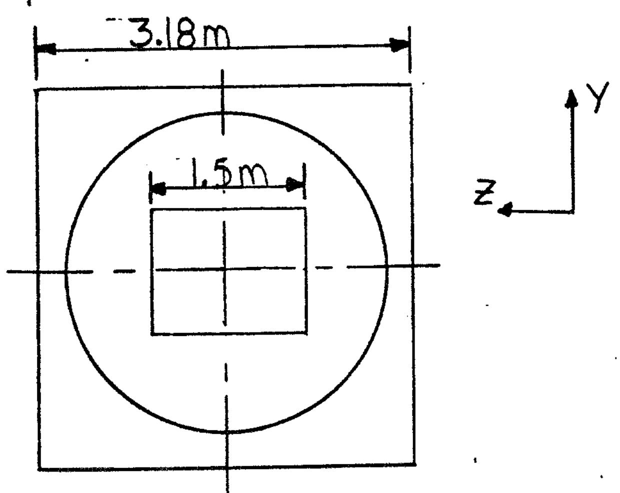


Figure 3-2

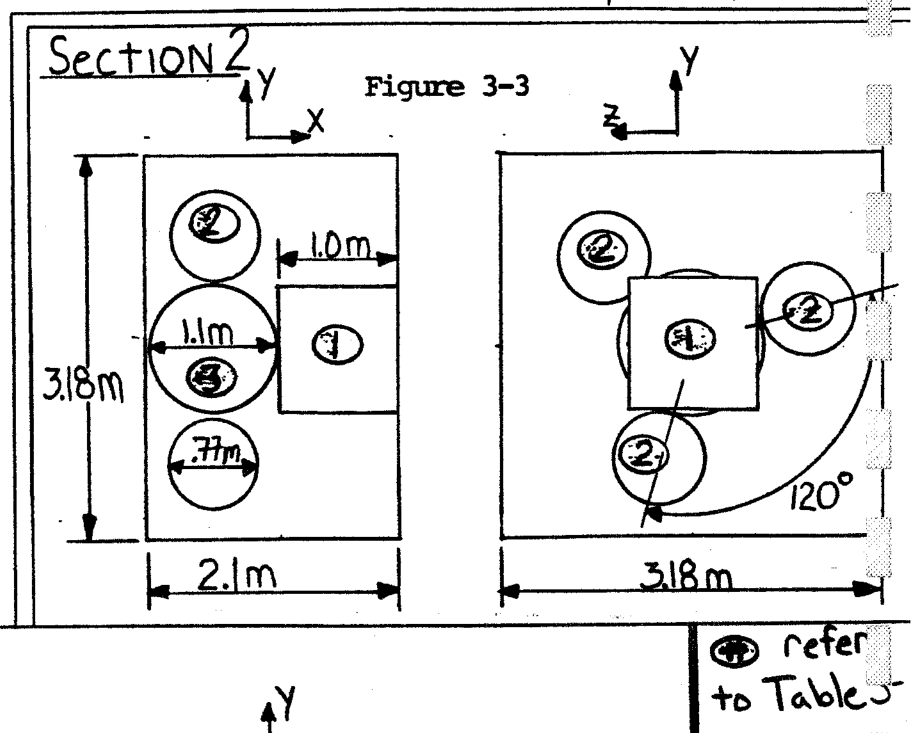


Figure 3-3

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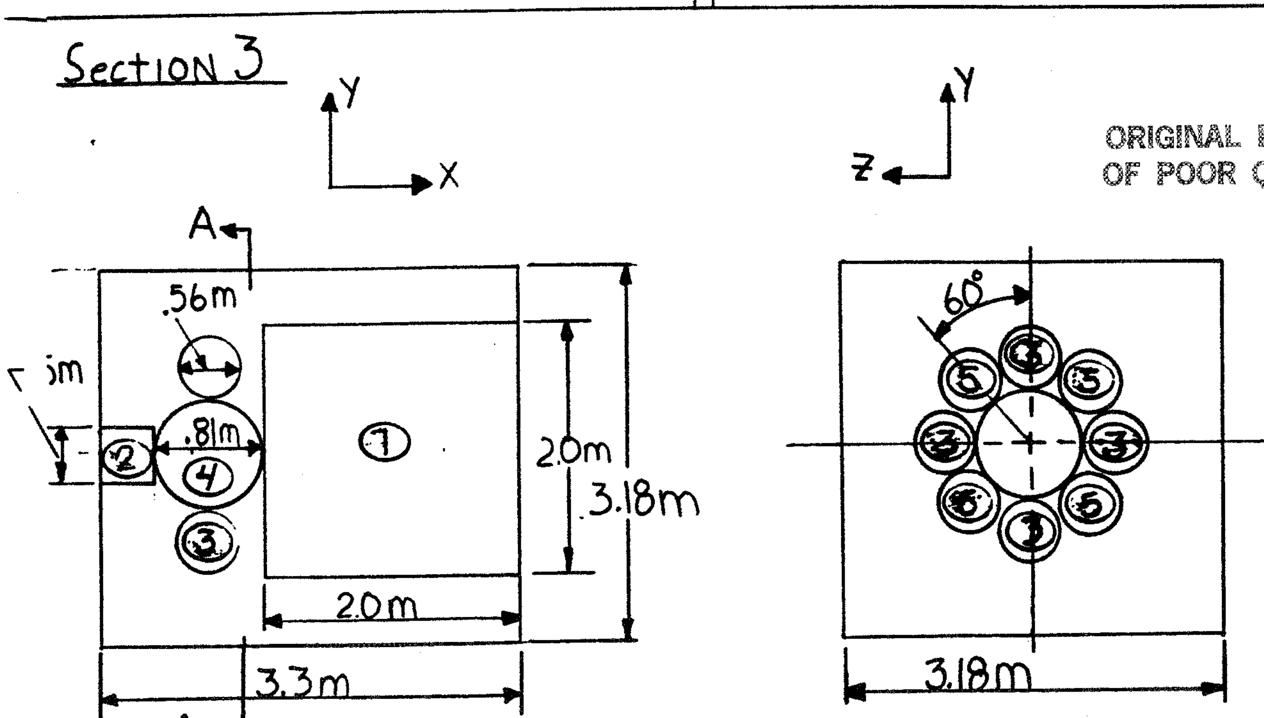

Figure 3-4

refer to Table 3-1

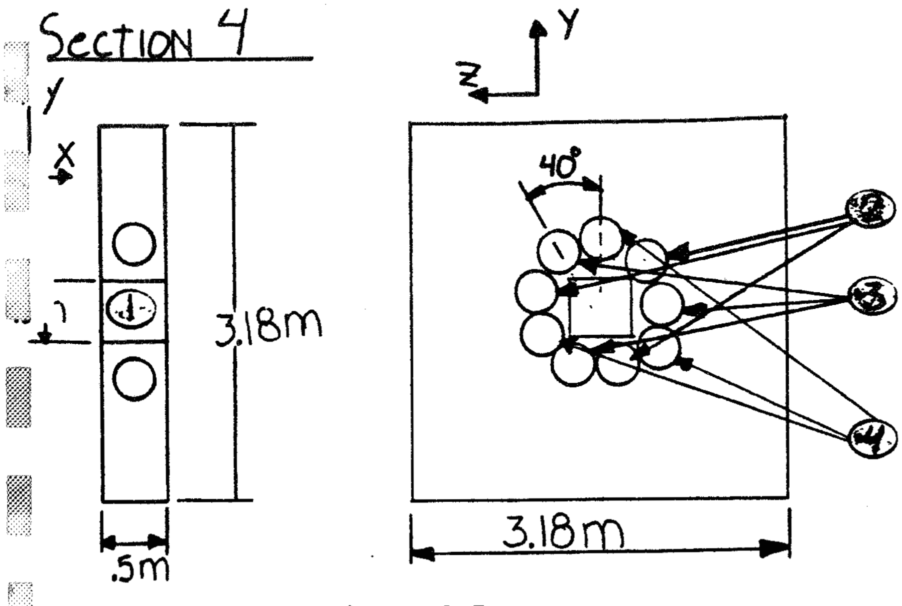

Figure 3-5

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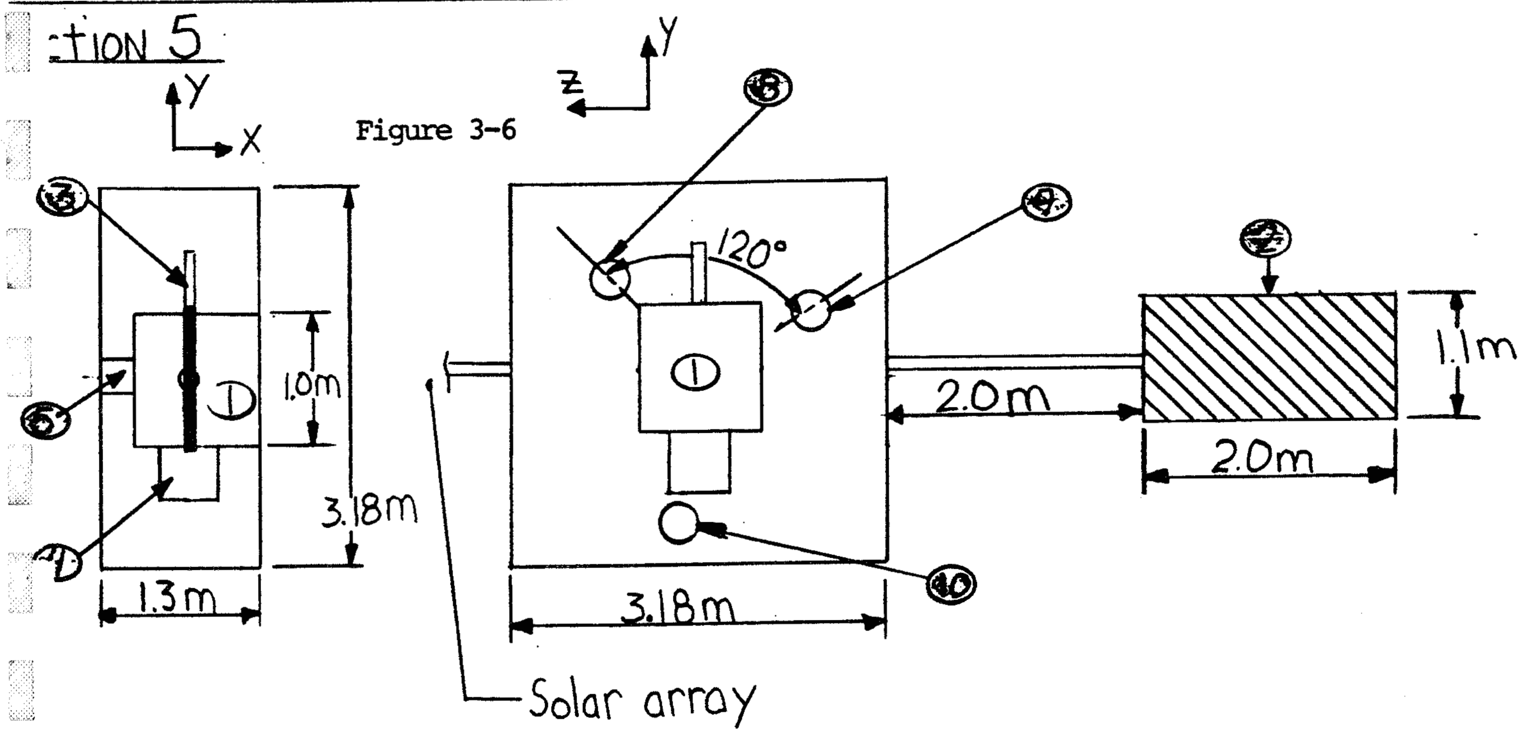

Figure 3-6

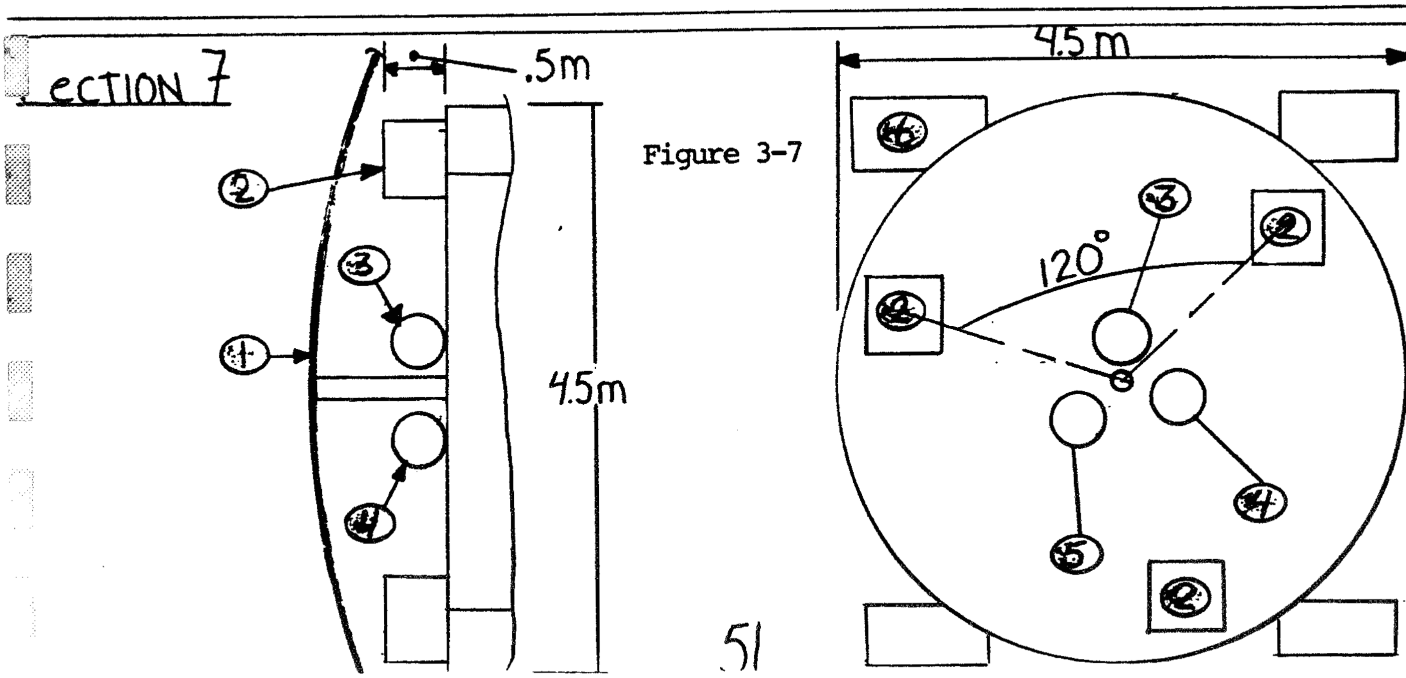

Figure 3-7

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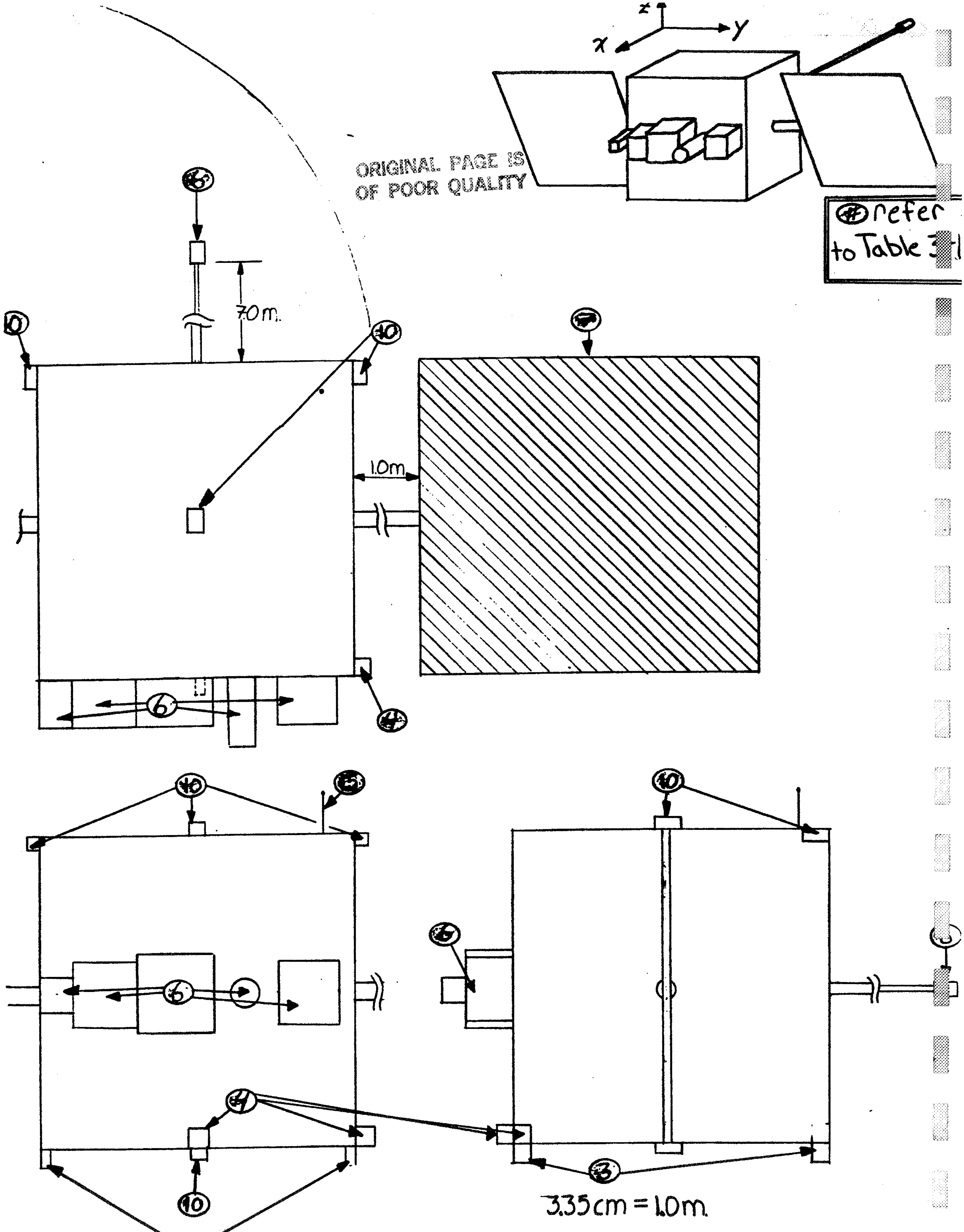

Figure 3-8

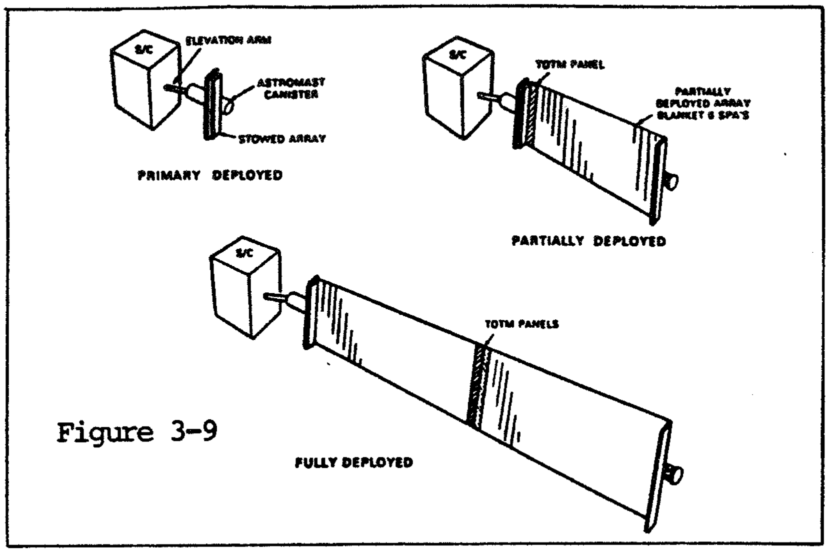

FIGURE 2 OLYMPUS ARRAY DEPLOYMENT SEQUENCE

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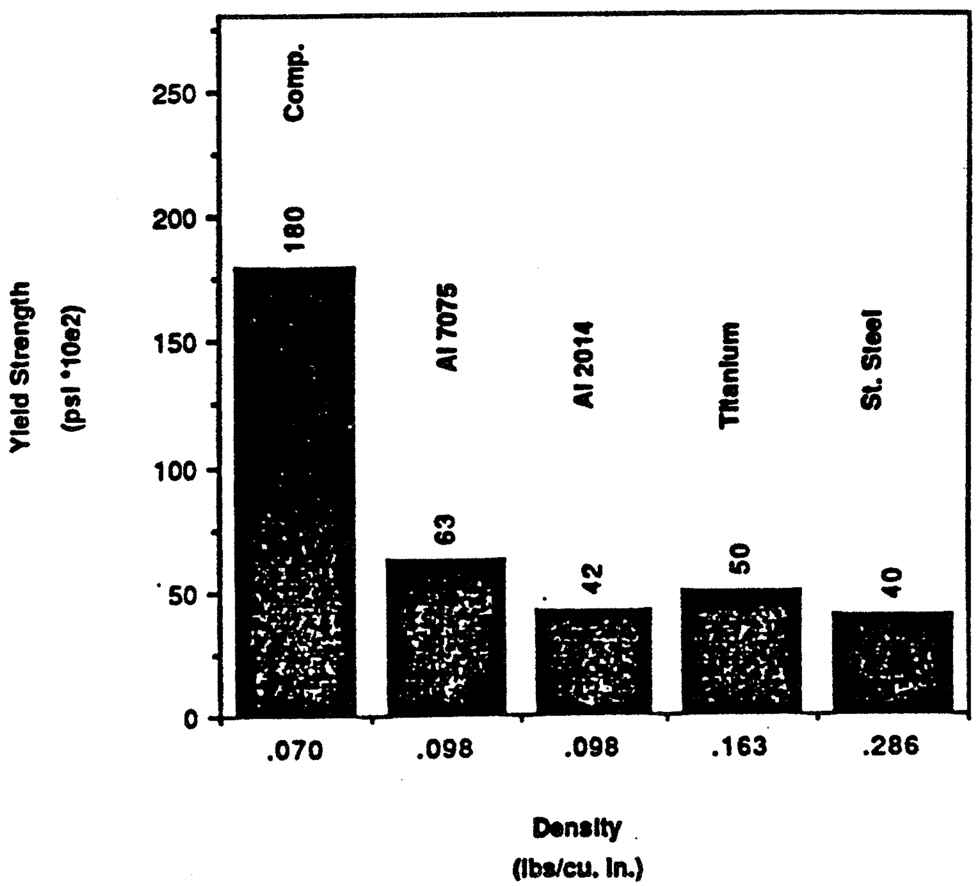
Figure 3-10 **Material Trade**



Attitude and Articulation Control Subsystem

An attitude and articulation control system is an integral part of any space mission, providing stability and directional or attitude control for spacecraft through various sensing and control systems. For MAVRIC, the proposed mission of a Mars aircraft delivery system, the attitude and articulation control system provides a very specific and crucial responsibility, allowing the delivery system to accurately and successfully deliver its payload to the Martian surface. This section will cover the analysis and design of such a control system for MAVRIC. Note that the terms "subsystem" and "AACS" are used in reference to the attitude and articulation control system.

Requirements

The Request for Proposal provides a number of system requirements that have direct bearing on the design of an effective attitude and articulation control system. The requirements that impact on the system design are: 1) a need to design a delivery system (spacecraft) to deliver a payload, initially a manned Mars aircraft, to the Martian surface by the first decade of the next century; 2) the design of two seperate systems in the spacecraft, a payload reentry vehicle and an orbiting instrument bus; 3) the design of a very flexible delivery system capable of carrying different payloads to different destinations; 4) the incorporation of currently available, "off-the-shelf" hardware whenever available; 5) a design that will have a minimum lifetime of four years; 6) a design that will incorporate the latest advances in artificial intelligence to enhance reliability and reduce costs; 7) an emphasis on simplicity, reliability, and minimum mass and cost; and 8) the need for four delivery systems, one to be used as a ground test system.

With these overall design requirements in mind, there is also a need to outline specific requirements for the attitude and articulation control system. These specific requirements include; 1) the need to send and to accept telemetry to and from the on-board command, control, and communications system. Inputs to the C^3 system include; a) inertial attitude and navigation information from gyros and accelerometers; b) star/sun/horizon sensor input for attitude reference or inertial navigation update; c) actuator position, such as that from a solar array gimbal; d) valve actuator positions from attitude control thrusters; and e) angular rate information from reactions wheels. Commands from C^3 include; a) adjustment or correction of spacecraft attitude through control system; b) actuator movement; c) valve actuation for thrusters or reaction wheel motor start-up or shut-down. Another specific requirement is: 2) the provision of electrical power through the power and propulsion subsystem.

Breakdown of the Design Process

The metholdology of the design process is crucial so that it be insured that the final subsystem design be adequately and effectively matched to the mission and its requirements. This requires that a step-by-step process be followed in order that one can adequately produce a design that completely fulfills what is required of it. This includes:

1. The identification and understanding of mission and subsystem requirements.
2. The identification and understanding of subsystem tasks and responsibilities.
3. Analysis of mission factors affecting performance or ability of subsystem.
4. Understanding of subsystem interaction with other subsystems.
5. Understanding and analysis of available subsystem options, as in component determination and ability sizing. Such an analysis would also include sensitivity and trade studies to most accurately gauge effectiveness of various options and consideration of requirements and necessary effectiveness of other subsystems.
6. When design approaches are finalized, subsystem performance and ability are matched to required functions and responsibilities, as determined by design requirements, other subsystem requirements, and factors affecting mission progression.
7. Simulation of subsystem operation to identify and correct potential problem areas or difficulties.

With this generalized approach, an effective and well-suited design can be obtained. Emphasis for this subsystem was not only placed on compliance with requirements, but also open and effective interaction with other subsystems to insure that smooth and efficient integration of systems resulted.

Subsystem Modal Operation

Specific subsystem operational parameters were established by identifying specific mission needs. Different modes of operation were designated to delineate varying subsystem operations throughout the mission. Six modes of operation were identified: 1) launch mode; 2) cruise mode; 3) planet capture mode; 4) aerobraking mode; 5) instrument bus orbit mode; and 6) reentry mode. In the launch mode, the attitude and articulation control subsystem provides correct spacecraft orientation for the delta-V burn which would place the spacecraft in a proper planetary transfer orbit to Mars. In cruise mode, the subsystem would insure spacecraft stability and proper attitude control during the transfer orbit by being able to detect and correct flight deviation or respond to spacecraft operation requirements such as power or thermal control needs through modification of spacecraft orientation. In planet capture mode, the subsystem would sense spacecraft position and provide the correct orientation and position information for a delta-V burn to terminate the planetary transfer orbit and initiate aerobraking. In the aerobraking mode, the subsystem would provide vehicle stability and attitude and position information during the aerobraking period. The instrument bus orbit mode involves the activation and operation of a seperate attitude and articulation control subsystem for the instrument bus, seperate from the payload reentry vehicle. The instrument bus attitude control system would provide the satellite with initial attitude information and control for orbit placement once it is released from the payload reentry vehicle, and then provide the necessary control for orbit stationkeeping and satellite function. In the reentry mode, the attitude control subsystem on-board the payload reentry vehicle would provide the necessary atttiude reference, guidance control, and stability for accurate reentry into the Martian atmosphere.

Subsystem Interaction

In the design process, it is imperative that open and complete communication exists between those subsystems involved in the design. This helps to insure that a design is well-integrated and subsystem components well-matched in providing complementary capability to assist or provide in fulfilling mission requirements. Below is a listing of MAVRIC design subsystem requirements and their interaction with the attitude and articulation control subsystem:

Mission Management, Planning and Costing -- Requires that this subsystem provide accurate spacecraft position and orientation through all stages of the mission, providing information for activation of rocket motors and other systems. Also, requires that AACS provide stability control of spacecraft throughout the mission. AACS requires, in turn, specific mission profile information to provide effective capability.

Structure -- Requires that AACS provide specifications of components to be used, namely mass and dimension information, for placement purposes. AACS requires that balance, mass property, and layout information on spacecraft(s) be provided for component placement and sizing.

Power and Propulsion -- Requires that AACS provide power requirements for components used. AACS requires that power availibility and fuel and thruster information be provided for component determination and sizing.

Science and Radio Relay Instrumentation -- Requires that AACS provide attitude control capability information for proper placement and choice of science instrumentation. AACS requires that pointing and scanning information be provided to properly designate components.

Command and Data Control -- Requires that AACS provide information on data processing and storage requirements, plus telemetry requirements. AACS requires that pointing and C^3 interface information be given for integration purposes.

Aerobraking -- Requires that AACS provide inertial guidance and stability and attitude control during aerobraking and reentry. AACS requires that aerobraking requirements be outlined as applicable to AACS.

Note there is a good deal of overlap and a great need for continuous and multi-level interaction between all subsystems.

Component Determination

Component determination is dependent upon mission factors that affect subsystem performance. These factors include environmental effects such as solar radiation torques and planet gravity gradients, and mission parameters such as spacecraft mass and size, component mass and placement restrictions, and attitude and articulation control requirements of other design subsystems. With these considerations in mind, along with the necessary interaction with other design subsystems, components can be designated and sized.

An attitude control system for the payload reentry vehicle and instrument bus was chosen. Several attitude control systems were considered for use for both spacecraft; dual-spin stabilization, active three-axis stabilization utilizing thrusters, and three-axis stabilization utilizing internal reaction wheels and thrusters. For the instrument bus, there was further consideration given to the number of reaction wheels used. These stabilization and control systems were determined by availability of current and anticipated technologies and and information from earlier space missions.

For the payload reentry vehicle, requirements for internal payload capability and aerobraking resulted in a three-axis active stabilization system for the designated attitude control system, it being deemed best suited for mission requirements. For the instrument bus, a three-axis reaction wheel stabilization system was designated, major arguments including a need for fine pointing and tracking of science instruments and the requirement for flexibility of maneuvering and orbit modification. Thrusters were also determined to be necessary for initial loading and periodic unloading of the reaction wheels, and possibly for those maneuvers requiring torques greater than that provided by the reaction wheels. For a more detailed trade analysis, please refer to Table 4-1.

With the attitude control system determined, proper sensing components were chosen for inertial attitude information and guidance along with active sensors to be used as attitude reference and inertial guidance update. Inertial navgation and guidance requires the use of gyros, which sense angular rate and angular rate integration along three-axes. Thus, gyros would be necessary for both the main payload reentry vehicle and the instrument bus.

Stabilization	Accuracy	Adaptability	Cost	Comments
3- Axis w/ thrusters	Accurate pointing capability	Adapts well to changing mission requirements	Very costly hardware	Provides autonomous control about all 3 axes Fuel requirements.
3-Axis w/reaction wheels	Very accurate pointing capability along all 3 axes. Much more accurate when compared to thrusters.	Same as above	Hardware is costly and complex.	Thrusters required for angular loading and unloading only. Precision pointing excellent.
Pitch Momentum Bias (1 pitch wheel)	Accurate rolll axis pointing, but yaw pointing through orbit coupling is poor.	Maneuverability is not as precise as other 2 systems	Not as costly as above	Advantage is lower weight and cost.

Table 4-1 Results of Stabilization System Trade Study

Component	Weight (kg)	Power Required (Watts)	Dimensions (m) or Volume (m^3)
Computer	8 kg	20 W	0.5 m square
Star Tracker (2)	1.5	1.5	0.064 Dia x 0.15 (add'l sunshield)
Gyro pack (3 x 2dof)	16.0	7.5	0.23 x 0.29 x 0.31
Thrusters (16)	1.0	6.5 (per pack of 4)	0.56 x 0.56 x 0.13 (per pack)
Fuel (and tank)	95	N/A	Oxidizer: 0.035 m^3 Fuel: 0.035 m^3 Pressurant: 0.0228 m^3

Table 4-2 AACS Components on Payload Reentry Vehicle

For attitude reference and inertial sensing update on the payload reentry vehicle, the use of star tracker-type sensors provide a very accurate indication of spacecraft position, and thus two of these sensors were designated. Information acquired with these sensors would not only provide navigation data and update, but also proper positioning of the solar arrays which are used to provide power to the spacecrafts. Specific details of sensor and gyro mounting in and on the payload reentry vehicle can be found in the Structures subsystem layout. Also, refer to Table 4-2 for a detailed listing of these components and specifications.

For the instrument bus, sensors are needed to supplement the gyro pack on board. These sensors are required to determine the necessary orbit orientation and attitude for use in maneuvering and stationkeeping. A pair of horizon sensors are required to accurately determine orientation along the pitch and roll axis, while the use of sun sensors will be needed to determine orientation about the yaw axis, as well as to provide reference information for accurate pointing of solar arrays (Note that all orientations are measured relative to the planet). Two sun sensors were deemed necessary as their placement on the satellite restrict each sensor's ability to detect the sun at certain orbit orientations. Again, a layout of these components can be found in the Structures subsystem section, and a listing of components and specifications can be found in Table 4-3.

For both the instrument bus and payload reentry vehicle, data processing and storage capability is provided by the C^3 subsystem. For an explanation of the handling of subsystem processing requirements, please refer to the C^3 subsystem section.

Component Specification and Sizing

Sizing of the thrusters used by the attitude control system aboard the payload reentry vehicle is dictated by the mass and size of the spacecraft and the requirements of thermal control and power generation, as well stability and guidance control during planet capture and aerobraking in the Martian atmosphere. Criteria established determined that a 360-degree rotation each day of cruise for the payload reentry vehicle would be a fair estimation to use for fuel and thrust requirements of the thrusters. Thus, the sixteen thrusters

Component	Weight (kg)	Power Required (Watts)	Dimensions (m) or Volume (m^3)
Computer	8 kg	20 W	0.5 m square
Sun Sensor (2)	0.5	0.3	0.107 x 0.066 x 0.114
Horizon Sensor (2)	1.0	3.5	0.10 x 0.50 x 0.10
Gyro pack (3 x 2dof)	16.0	7.5	0.23 x 0.29 x 0.31
Thrusters (16)	1.0	3.75 (per pack of 4)	0.37 x 0.37 x 0.06 (per pack)
Fuel (and tank)	16	N/A	Oxidizer: 5.85×10^{-3} m^3 Fuel: 6.1×10^{-3} m^3 Pressurant: 0.54 m^3
Reaction Wheel (3)	6.8	15.0	0.35 Dia. x 0.15

Table 4-3 AACS Components on Orbiting Instrument Bus

used, twelve required for attitude control about all axes and four for redundancy or maneuver capability, were sized and their fuel requirements were determined by this estimation and the additional requirement of stability during mission cruise, planet capture and aerobraking. The thrusters designated produce 20 Newtons of thrust each. It is estimated that the fuel requirements of the spacecraft thrusters would be approximately 95 kg of bipropellant for the entire mission. Refer to Table 4-2 for required masses, volumes, and tank sizes of fuel components and for the estimated size, mass, and power requirements for the other components. Also, please refer to Appendix 4A for equations and formulas used for the estimation. For a more detailed explanation of the bipropellant and sizing equations, please refer to the Power and Propulsion Subsystem section.

Sizing of attitude control components on the instrument bus is dictated by its mass and size, its mission requirements, and anticipated destabilizing and disturbing effects from environmental or other external sources. Specifically, the attitude control system is required to perform stationkeeping in its designated orbit, with its capability to stabilize the instrument bus from disturbing effects, to perform orbital modifications if and when needed, and to provide the necessary pointing information and guidance for the specified science instrumentation and other components, such as solar array gimbals and communication antennae orientation.

It was decided to employ a reaction wheel system as opposed to a three-axis system using thrusters because of the need for accurate pointing and scanning requirements of the science instrumentation on-board the instrument bus. It was decided that for reliability and simplicity reasons the science instrumetation be mounted directly on one of the faces of the instrument bus structure. A three-axis system employing thrusters is not capable of providing accurate pointing and scanning to the degree required by the science instrumentation, unless one employs a gimballed scanning platform mounted on the instrument bus, an option discarded for reasons stated above. Also, it was decided to employ three reaction wheels instead of a single speed-biased pitch wheel (pitch momentum bias system). Although a single pitch wheel could have fulfilled the necessary pointing and scanning requirements along the roll and yaw axes due to rate coupling, it was determined that fine and accurate positioning about **all** axes was necessary for accurate data gathering, something that a single pitch momentum wheel could not provide. Thus the use of a three-axis system with three reaction wheels mounted about each of the principal axes (pitch/roll/yaw) was determined to be necessary.

Using the equations of motion (linearized) along with those equations representing solar radiation torques and gravity gradients, an estimation of reaction wheel and thruster requirements are obtained. Requirements dictate that reaction wheels not only provide accurate pointing and scanning capability by reorienting the instrument bus, but also that they be able to provide adequate attitude and orbit correction capability (stationkeeping) from the highest predicted disturbing effects (torques), which are assumed to be those created by solar radiation and gravity gradients. Also, the thrusters must be able to provide the capability to load and unload angular momentum from the reaction wheels, as well as provide additional or redundant attitude control and maneuvering.

The designated reaction wheels are designed to provide 0.5 N-m of thrust each with maximum angular momentum of 20 N-m-sec for each wheel, with ample capability to counteract and correct for the aforementioned disturbing torques. The sixteen thrusters designated provide full rotation about all axes, with each thruster generating 1 Newton of thrust. Twelve thrusters are required for three-axis capability, with four needed for redundancy or possible maneuvering requirements. Refer to Appendix 4B for a listing and method of equations used, and to the Structures subsystem section for component layout and placement. Also, refer to Table 4-3 for component listing and specifications.

Problem Areas

As is expected with all design processes, there were a number of problems that arose. The most major of these was optimal placement of attitude and articulation control components on the spacecraft components, i.e. the payload reentry vehicle and the instrument bus. Because of the need for proper balancing and other subsystem component requirements, there were compromises made in the placement of some components of this subsystem, as there were with components with other subsystems. For example, attitude reference sensors require an unobstructed field of view (within each sensor's viewing parameters), which was not always completely possible, thus requiring compromise with other subsystems. Also, thruster placement had to be chosen carefully as to avoid possible distortion of science instrument viewers or actual physical damage to these instruments or other spacecraft components, yet without affecting the performance of the thruster system. A problem area yet to be resolved is the accurate estimation of attitude control thruster requirements during aerobraking, of which no applicable information or techniques could be provided.

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Appendix 4A

Inertia Matrix of Spacecraft: $\begin{Bmatrix} I_{xx} \\ I_{yy} \\ I_{zz} \end{Bmatrix} = I$

$$\tau_x = I_{xx}\dot{\omega}_x$$

Torque on spacecraft: $\vec{\tau} = I\dot{\vec{\omega}} \longrightarrow \tau_y = I_{yy}\dot{\omega}_y$

$$\tau_x = I_{zz}\dot{\omega}_z$$

Assume angular rates are zero $(\vec{\omega} = 0)$

Desired angular rate of $^s/_c$ when turning: ω_x = 1 deg/sec = 0.0157 rad/se

$$\Delta t = \Delta\omega_x/\dot{\omega}_x \dashrightarrow \omega_x(0) = 0$$

Obtain Δt (τ is known from given thrust and lever arm of thruster)

Mass of fuel used = $M\Delta t/I_{sp}g_o\ell$ where

M = thruster moment

I_{sp} = specific impulse of fuel

g_o = graviation acceleration Earth at sea level

ℓ = lever arm of thruster

= $F\Delta t/I_{sp}g_o$

F = thrust force

Δt = burn time

($M\Delta t$ = angular momentum = h)

Result is mass of fuel used in each impulse to accelerate the spacecraft from rest to a constant angular rate of 1 deg/sec, or to decelerate the spacecraft from this angular rate to zero.

Multiply this result by 2 to find fuel needed for a rotation manuever, and then multiply by 211 (days) to find the total mass of fuel. Add estimation factor for fuel consumed during aerobraking

Refer to Power and Propulsion Subsystem for propellant and tank sizing equations.

Appendix 4B

Equations of Motion of Instrument Bus in orbit:

$$\vec{\tau} = I\dot{\vec{\omega}} + I\dot{\vec{\omega}}^w + [\dot{\vec{\omega}} \times (I\vec{\omega} + I^w\vec{\omega}^w)$$

where $I = \begin{Bmatrix} I_{xx} \\ I_{yy} \\ I_{zz} \end{Bmatrix}$ --> Inertia matrix of the instrument bus

$I^w = \begin{Bmatrix} I_{xx}^w \\ I_{yy}^w \\ I_{zz}^w \end{Bmatrix}$ --> Inertia matrix of the reaction wheels along each principal axis.

$\vec{\omega} = \begin{Bmatrix} \dot{\phi} \\ \dot{\theta} \\ \dot{\psi} \end{Bmatrix}$ --> Angular rates about roll, pitch, and yaw axes $(\hat{e}_1, \hat{e}_2, \hat{e}_3)$.

$\vec{\omega}^w = \begin{Bmatrix} \omega_x^w \\ \omega_y^w \\ \omega_z^w \end{Bmatrix}$ --> Angular rates of reaction wheels about their respective axes.

Solar radiation pressure:

For perfect absorber (blackbody) $= 4.5 \times 10^{-6}/l^2$ N/m^2
For perfect absorber $= 9.0 \times 10^{-6}/l^2$ N/m^2

where l = distance from Mars to the sun in a.u. (1.52 a.u.)

Estimate center of pressure on instrument bus -- Assume average 6 cm off center of mass.Find solar radiation torque on instrument bus.

$\tau_{sr}\Delta t_{sr} = \Delta h$ -->where h_{max} =20 N-m-sec
~~h_{avg} =10 N-m-sec~~
Δh =10 N-m-sec

Δt_{sr} derived from constant angular acceleration
τ_{sr} is the solar radiation torque

Obtain Δt for thruster burn with similar equation as above using torque of the thruster and Δh obtained above.

Compute fuel mass from equation: $M\Delta t/I_{sp}g_o l$

Gravity-gradient torques (linearized) -- for circular orbits

$$\tau_x = -3\omega_o^2(I_{yy} - I_{zz})\phi$$
$$\tau_y = -3\omega_o^2(I_{xx} - I_{zz})\theta$$
$$\tau_z = 0$$

Find fuel masses using equations of motion and fuel mass equation given above. Assume no initial angular rates of instrument bus.

COMMAND and DATA CONTROL

by Peter Hjellming

INTRODUCTION

As noted in the Mission Planning section, we have decided to emphasize the communication support aspect of the first mission. Although this choice decreases the importance of the data coming from the science instruments, the workload on the CDC is not decreased because of the increased communication requirements. The CDC sub-system was designed to fulfill theses varied objectives using a structure method of attack. I started by pulling the CDC requirements from the RFP. I also outlined what the satellite would have to do during different phases of the mission. I then compiled the design objectives as stated in the RFP and listed the assumptions I was making in formulating my design. With all this information I went about choosing my systems that would satisfy my requirements and justifying their choice.

REQUIREMENTS

The tasks that the CDC sub-system performs can be divided into internal concerns, and external concerns. Internally the CDC takes care of integrating and commanding the various parts of the spacecraft. Externally the CDC relays information from various sources to it intended destination.

Internal Requirements:
- Collect telemetry from all sub-systems.
- Generate commands based on telemetry.
- Send telemetry to Ground Control.
- Receive commands from Ground Control.
- Send commands to sub-systems.

Get necessary power from PPS sub-system.

External Requirements:
- Collect science instrument data.
- Decide if landing site is clear.
- Send science instrument data to Ground Control.
- Receive communications from Mars Station and Aircraft.
- Relay communications to Mars Station and Aircraft.

Mission Profile:

Pre-Launch:	Interface and test all subsystems.
Launch:	AACS telemetry on progress of the burn. Commands to PPS throughout burn.
Flight:	AACS telemetry for in-flight corrections. Monitor sub-systems, relay telemetry to Ground Control.
Approach:	Prepare for aerobraking.
Aerobrake:	AACS interactions to control aerobraking.
Touchdown:	Evaluate landing site. Control descent into atmosphere.
Instr. Bus:	Begin science mission. Relay station/plane communications.

DESIGN

There were many requirements in the RFP that restricted the design of the CDC sub-system. The one which we have emphasized specifically is simplicity. This is further encouraged by other requirements. All equipment used by the CDC had to be available on or before 1998 and where possible off-the-shelf technology was to be used. The lifetime of the sub-system had to be at least four years. The use of artificial intelligence in the design and operation of the CDC was highly encourage to minimize control from the ground. The spacecraft would be launched into low Earth orbit by the Space Shuttle. Finally, the design had to be capable of a variety of missions.

With these design goals in mind and the requirements listed above, thought could be given to performing the mission. But some assumptions had to be made before the design could be completed. As mentioned earlier, we decided to emphasize the communication support aspect of this mission. As a means of decreasing the power, antenna pointing, and data storage requirements of the CDC it was decided to send the instrument bus telemetry to Ground Control through the Mars Station after the bus is in orbit around Mars. The Mars Station will have a more capable CDC sub-system than the bus and the station could analyze the data it needs immediately, such as weather information.

INTERNAL SYSTEMS

Internal systems are systems which are used to meet one of the internal requirements listed above. The three components of this system are the spacecraft interface unit, digital computer, data storage, and S-band radio.

The spacecraft interface unit collects the telemetry from all the sub-systems, sends it to the computer, and distributes the commands that the computer comes up with. It also collects the power for the CDC components.

The digital computer generates on-board commands for the instrument bus. It uses telemetry gathered by the spacecraft interface unit and algorithms stored in the data storage to generate commands and relieve Ground Control of as much work as possible.

The S-Band radio and antenna will be used to send telemetry and receive commands from Ground Control. The radio will be transmitting at enough power to allow 4.0-E16 watts of power to be received by the DSN. This amount of power allows for high data rate bursts and can be used for compensating for atmospheric conditions. The burst transmissions help decrease the amount of time that the DSN is occupied with our particular satellite. Figure 3-1 shows the configuration of the antenna. It is rigidly mounted to the spacecraft. Pointing is performed by rotating the spacecraft and point the antenna to Earth during transmission and reception. Figure 5-1 is a graph of S-Band Antenna size vs. Tranmitting Power. Figure 5-2 shows that the received power will allow for roughly 250 kilo-bits per second of data to be transmitted. This is well within the amount of telemetry the spacecraft will be collecting about itself. As mentioned above, no direct communication with Earth will be necessary once at Mars. Therefore the S-Band antenna will be discarded on arrival.

EXTERNAL SYSTEMS

External systems satisfy external requirements. The components used are the mission interface unit, digital computer, data storage, and the X-Band radio.

The mission interface unit collects the science instrument data and packages it into a transmittable form. It is very similar to the spacecraft interface unit except for being dedicated to the science instruments.

The digital computer is used to analyze the data and evaluate the condition of the landing site of the payload.

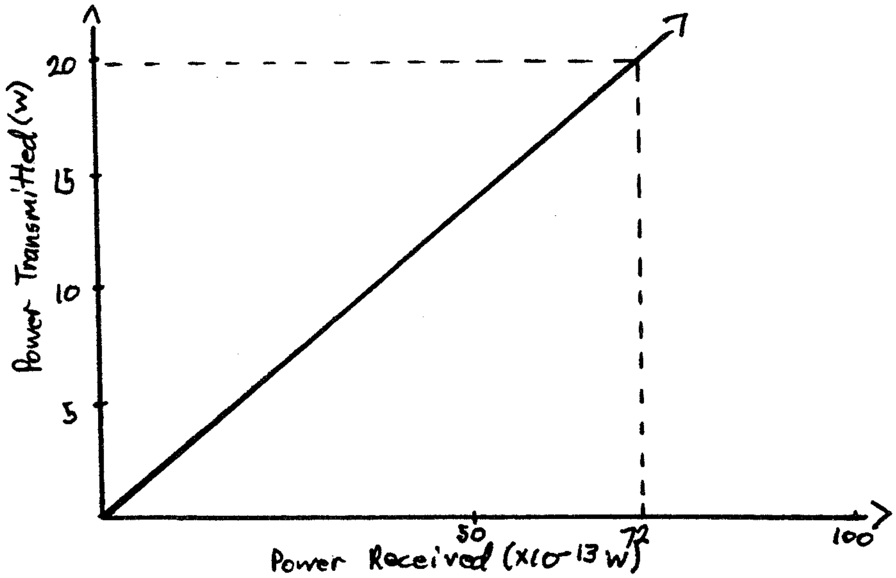
To do this it uses algorithms stored in data storage. In addition to holding algorithms, the data storage also stores telemetry that is waiting to be transmitted. This is particularly important if the Mars Station is obscured by dust storms. It also allows for burst transmissions over a short period of time. This would minimize the load on the Station due to the instrument bus.

The X-Band radio will be used to communicate with the surface of Mars. To eliminate the need for pointing and simplify placement, an isotropic antenna will be used. As Figure 5-3 shows the relationship between the power received by the Mars station and the power tranmitted by the satellite is a linear one. A 20 watt transmitter is easily attainable and would resulted in the highest data rate. Figure 5-4 shows that with the Mars Station receiving 7.26-E12 watts almost half of a Giga-bit could be transmitted in a second. Such high data rates would provide for fast transmission of imaging pictures.

CONCLUSION

Although the CDC sub-system may have been able to save some weight by being designed for only this particular mission, the RFP specifically stated that a variety of missions might be required. This sub-system has the ability to meet the requirements of missions that have longer communication distances and greater data rates. A summary of the CDC components, their weight, and the power they require at various times in included in figure 5-5.

Figure 5-3
X-Band Transmission Power
vs
Power Received at Mars



$$P_r = \frac{P_t}{4\pi D^2}$$

$$D = 1.7+E06 \text{ m}$$

$$P_r = 3.63-E13 * P_t$$

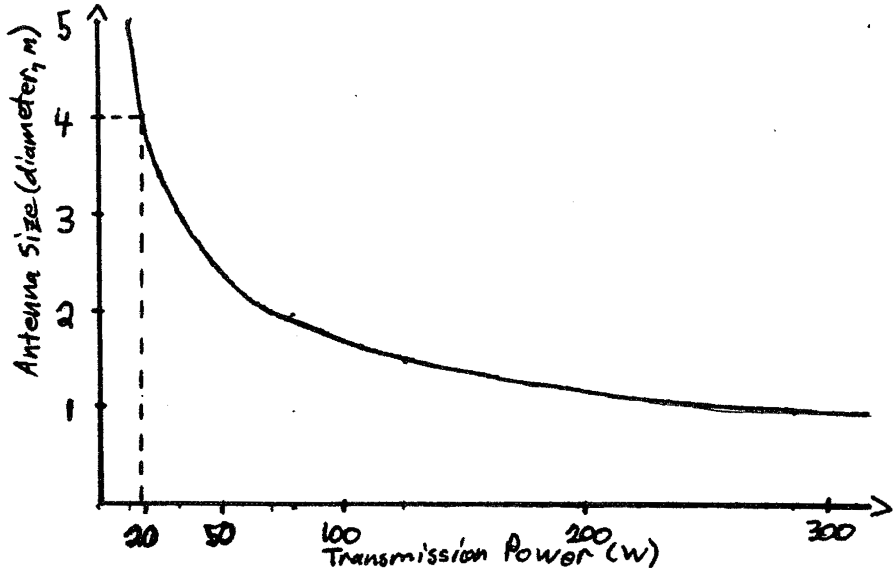
Figure 5-4
X-Band Data Rate

$$B = \frac{P_r}{kT(SNR)} \log_2(SNR+1)$$

P_r = 7.26-E12 W
k = 1.38-E23 J/K
T = 25 K
SNR = 20

B = 4.62+E09 bits/second

Figure 5-1
S-Band Antenna Size
vs
Transmission Power



$$d_t = \frac{4CD}{fzd_r\pi}\sqrt{\frac{P_r}{Pt}}$$

P_r = 4-E16 W
C = 3+E08 m
D = 1.79+E11 m
f = 2.295+E09 /s
z = 0.55
d_r = 64 m

$$P_t = \frac{286.54}{d_t^2}$$

Figure 5-2
S-Band Data Rate

$$B = \frac{P_r}{kT(SNR)} \log_2(SNR+1)$$

P_r = 4-E16 W
k = 1.38-E23 J/K
T = 25 K
SNR = 20

B = 2.55+E05 bits/second

Figure 5-5
CDC Component Information

Component	Mass(kg)	Power(W)
Spacecraft Interface Unit	5	5
Computer	8	10
Data Storage	9	10
S-Band Radios	10	20
S-Band Antenna	18	-
Mission Interface Unit	3	5
X-Band Radio(2)	5	20

Power Needed:	Silent	Transmitting
In flight	30	50
Aerobraking	25	-
In Orbit	30	50

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SCIENCE INSTRUMENTATION

MISSION SCIENCE OBJECTIVES FOR MAVRIC

The scientific mission of the satellite is:

1) To determine the origin, evolution, and present state of the solar system.
2) To better understand the Earth through comparative planetary studies.
3) To understand the relation ship between the chemical and physical evolution of the solar system and the appearance of life.

The above goals are to be achieved by:

1) Determining the elemental and mineralogical character of the Martian surface on a global basis.
2) Determine the distribution, abundance, sources, and concentrations of volatile materials and dust.
3) Define the global gravitational field.
4) Measure the global topography.
5) Explore the atmospheric structure and its circulation in detail.
6) Establish the nature of the Martian magnetic field.

Additional overall project requirements directly influencing the science instrumentation subtask include:

1) The design will stress simplicity, reliability, and low cost.
2) Performance, weight, and cost should be optimized in design tradeoffs.
3) The spacecraft should use off-the-shelf hardware where available, but should not use materials or techniques expected to be available after 1998.
4) The spacecraft will have a design lifetime of four years, but nothing in its design should preclude it from exceeding this lifetime.
5) Nothing in the spacecraft's design should preclude it from performing several possible missions.

The MAVRIC design team has chosen to implement these objectives and requirements so as to maximize aircraft support. Forecasting weather conditions that the aircraft might encounter and supplying navigational aid to the aircraft during its operation have been deemed vital for the success of the Manned Mars Aircraft. This choice of accent on the design philosophy has several direct effects on the science instrumentation subtask, the discussion of which follows.

SELECTION OF INSTRUMENTS

The method of attack for selecting scientific instruments was to perform library research to find information on missions with similar scientific objectives. Analysis included past missions such as Mariner, Viking and Pioneer, present missions such as Voyager, Galileo and Earth satellites, and future missions such as the Mars Geoscience Climatology Orbiter. Several questions were then asked. What hardware and techniques used on these missions can be applied to MAVRIC? What changes and modifications would need to be made to suit our needs? What technology can be incorporated that will be available by 1998? Is there anything that we will need to design from scratch? Applicable instruments were then chosen based on performance, weight and cost tradeoffs. The following instruments were selected to be carried by the MAVRIC satellite. Listed with each instrument are the mission science objectives they fulfill.

1) Ultraviolet-Visual-Infrared Spectrometer
   This instrument will determine elemental and mineralogical character of the Martian surface, map concentrations of water and carbon dioxide both in the atmosphere and on the surface, and determine the concentration of ozone in the Martian atmosphere.

2) Gamma Ray Spectrometer
   Measures abundance of elements, volatile materials and dust on Mars' surface.

3) Magnetometer
   Will establish the nature of Mars' magnetic field.

4) Radar Mapper
   Will be used to map Mars' topography.

5) Ultraviolet Photometer
   Detects atomic hydrogen and will explore atmospheric structure.

6) Radio Science
   The satellite radio relay and Mars base radio link will be used to define Mars' gravitational field.

7) Imaging
   The imaging system will analyze weather patterns including atmospheric circulation and will be used for forecasting weather conditions that the Manned Mars Aircraft may encounter.

All of these instruments have a proven history of ground and flight test. For details concerning instrument power requirements, data rates, weights and sizes see figure 1.

PLACEMENT OF INSTRUMENTS

All instruments are placed on one face of the cubic satellite bus. This configuration was chosen since the satellite will be in a synchronous orbit over the Mars base longitude and one face will remain pointed at the planet throughout the orbit (for details concerning satellite orbit see the Mission Planning section). The instruments are placed symmetrically about the satellite center of gravity in order to simplify the satellite inertia tensor (for details of instrument placement effect on the satellite inertia tensor as calculated using INERT see the STRUCTURE section). See figure 2 for details of instrument placement.

SCANNING/POINTING REQUIREMENTS

As stated above all instruments are placed on one face of the satellite bus and will remain pointing at the planet throughout the satellite orbit. From the satellite's synchronous orbit a field of view of 18.79 degrees is required to collect data from the entire planet. All instruments can collect data from this field range. In addition, the imaging system is capable of scanning the width of the planet for narrower, more detailed information. For details of instrument fields of view and scanning abilities see figure 1. The satellite will be kept pointing at the planet by the attitude control gyros. For details of the gyros and stability requirements see the Attitude Control section. The only exception to the above methodology will be after aerobraking when the satellite will temporarily scan the landing site from a low Mars orbit (to insure safe landing conditions for the aircraft payload) before boosting itself into a synchronous orbit over the Mars base longitude. For details of these events see the Mission Planning section.

INTERACTIONS

1) Mission Planning
   Mission objectives and science objectives were analyzed and found to be contradictory. Aircraft support was given first priority and thus a synchronous orbit over the Mars base longitude was chosen.
2) Structure
   Instrument weight was minimized since this directly influenced material costs and fuel needs. Instruments were placed symmetrically about the satellite center of mass so as to simplify the satellite inertia tensor.
3) Aerobrake
   Instrument G-load and temperature tolerances are not exceeded during the aerobraking maneuver. See Aerobrake section for details.
4) Attitude Control
   Instrument pointing requirements are met by the attitude control gyros. Scanning is only necessary for the imaging system thus all instruments have been placed directly on the satellite bus rather than on a scan platform. See the attitude control section for details.
5) Power and Propulsion
   Science Instrumentation power needs are met by Power

and Propulsion. See Power and Propulsion section and figure 1 for details.

6) Command and Data Control
Instrument control commands are received from the Mars base and relayed to the science instruments by CDC. Data collected by the science instruments is converted from analog to digital format and relayed to CDC for recording or transmission to the Mars base. For details see the Command and Data Control section and figure 1.

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FIGURE 6.1

N TRUMENTS	MASS (kg)	POWER (WATTS)	SIZE (cm)	DATA RATE (bps)	SCAN RANGE (deg)	FIELD OF VIEW (deg)
N-VIS-IR F CTROMETER	10	8	40×40×30	100,000	—	45
AMMA RAY E CTROMETER	15	14	40×40×30	1300	—	45
AGNETOMETER	7	7	10dia × 15	100	—	3-AXIS 360
RADAR MAPPER	30	20	50×50×30	100,000	—	45
L RAVIOLET PHOTOMETER	5	8	20×20×30	20 kbits per week	—	20
ADIO SCIENCE	SEE DATA	COMMAND CONTROL	AND SECTION		—	—
I. IAGING	30	18	20dia × 45	100,000	90	30
OTAL	97	75	—	301,400	—	—

NOTE:

INSTRUMENT MASSES INCLUDE ELECTRONIC EQUIPMENT NECESSARY FOR ANALOG-DIGITAL CONVERSION AND TRANSMISSION TO COMMAND AND DATA CONTROL, THERMAL PROTECTION AND HEATERS AROUND TELESCOPE TUBES ON IMAGING CAMERA.

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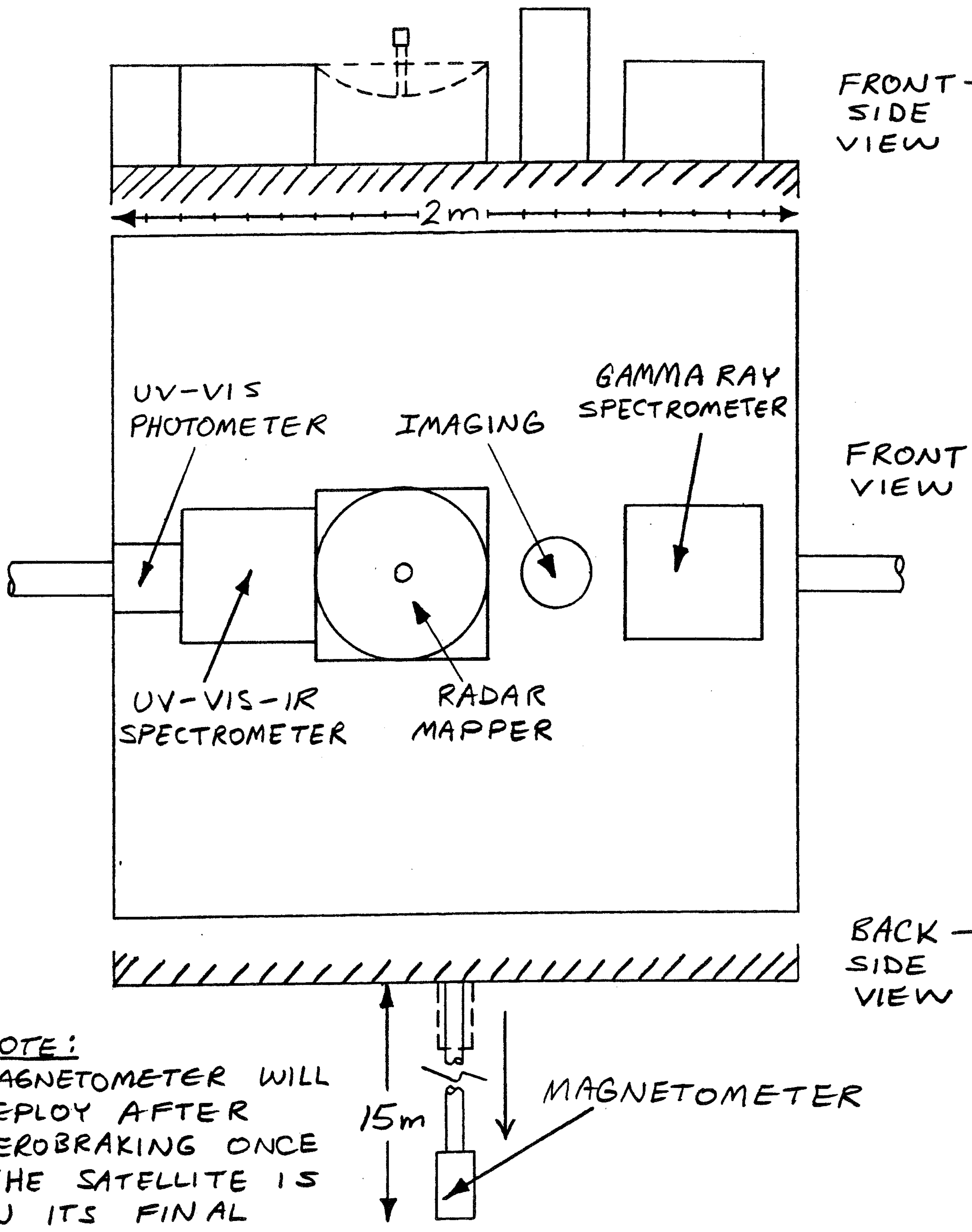
FIGURE 6.2



NOTE:
MAGNETOMETER WILL DEPLOY AFTER AEROBRAKING ONCE THE SATELLITE IS IN ITS FINAL SYNCHRONOUS ORBIT.

Distillation of RFP Requirements

Examination of the RFP provides the following requirements which apply directly to the Power and Propulsion Subsystem (PPS) design:

1. The PPS should use off-the-shelf hardware where available.

2. The PPS should not use materials or techniques expected to be available after 1998.

3. The PPS will have a design lifetime of four years, but nothing in its design should preclude it from exceeding this lifetime.

4. Nothing in the PPS design should preclude it from performing several possible missions, including sample return or multiple body fly-bys.

5. The spacecraft and instrument bus will both require separate PPS.

6. The design will stress simplicity, reliability, and low cost.

7. Performance, weight and cost should be optimized throughout the design.

Additional general requirements for the PPS include:

8. The PPS must be able to send telemetry to the C3 subsystem.

9. The PPS must be able to accept commands from the C3 subsystem.

10. The PPS must be self-powered.

11. The PPS must have some form of control outputs, such as power relays.

12. The PPS must have some form of sensor input, such as temperature sensors.

13. The PPS must provide an uninterrupted power supply.

14. The PPS must be protected from load faults and outages.

15. The PPS must allow for monitoring by the mission support team.

POWER SYSTEM

This mission requires two separate power systems, one for the spacecraft, and one for the instrument bus (Satellite). The following tables illustrate the power requirements of the two vehicles at different times in the mission profile.

Spacecraft	Power Required (Watts)		
	Cruise	Aerobraking	Parked Martian Orbit
AACS	100	62.5	100
C3	50	20	50
Contingency	20	17.5	20
TOTAL	170	100	170

Instrument Bus (Satellite)	Power Required (Watts)		
	Cruise	Aerobraking	Deployment
AACS	--	--	160
C3	--	--	50
Science	--	--	75
Contingency	--	--	25
TOTAL	--	--	310

Two power systems for this mission were considered: 1) solar arrays augmented with batteries or fuel cells to maintain power during eclipse periods of the mission, and 2) Radioisotope Thermoelectric Generators (RTGs). There were three major reasons why Solar Arrays have been selected over RTGs for this mission. The first reason is shown in figure 7-1 (Reference 5). This diagram illustrates for distances from the sun of 3.5 AU (AU = distance from the sun to the earth) and closer, a solar array system augmented with a 20 kg fuel cell has a better power per kg mass ratio (Specific Power) than does the most advanced RTG to date. Another factor that lead to this decision is the fact that RTGs are nuclear power sources which could be environmentally hazardous. It is very difficult to obtain clearance to use RTG power systems on spacecraft today. A final major disadvantage is that shielding is needed to isolate the radioactive power source of RTGs from sensitive instruments. This leads to extra weight and restricts placement of other spacecraft components.

For the solar arrays, two systems to augment the arrays during eclipse were studied, 1) batteries and 2) regenerative fuel cells. Two types of batteries were examined, Ni-Cd and Ni-H2 batteries. Because of long eclipse times associated with this mission (11.25 hrs for the instrument bus, and 3 days for the spacecraft (see mission planning section) weight of the system is going to be of upmost importance. Ni-Cd and Ni-H2 have similar performance capabilities, although Ni-H2 batteries have not been mission tested yet where as Ni-H2 batteries have over 20 years flight history. The most significant difference between the two is Ni-Cd batteries are generally 3 times the mass of equivalent Ni-H2 batteries (Reference 3), so it would be to this missions advantage to utilize the Ni-H2 battery system. Ni-H2 batteries are relatively new technology but they are planned to be used on the space station (Reference 3) so they will be available before this mission gets off the ground.

The other power source considered is the LH2/LOX fuel cell. LH2/LOX fuel cells have been used for spacecraft power systems since the 1960's Gemini program. There are three LH2/LOX fuel cells used on the Space Shuttle. It is proven technology. The only real determining factor on which system (batteries or fuel cells) to use, is unit weight. As shown in figure 7-2, the length of eclipse time is the primary concern when looking at what type of system to use (weight is the primary factor in system selection). For an eclipse time of more than 9 hours, at 310 Watts power, it is optimal to use fuel cells. The instrument bus is going to experience an 11.25 hour eclipse every 24.72 hours with a required load of 310 Watts. It is evident that fuel cells should be used in this case. To stress the point, the following are calculations of the battery weight required compared to the fuel cell weight required to maintain instrument bus power for 11.25 hours.

<u>Ni-H2 Battery</u> (Reference 3)

PL = Power Load = 310 W
Te = Time of Eclipse = 11.25 hrs
DOD = Depth of Discharge = 55%
WHrs/Kg = 75 WHrs/Kg

Stored Energy = PL*Te/DOD = (310 W)*(11.25 hrs)/.55 = 6341 WHrs

Weight (Battery) = (Stored Energy)/[(WHrs/Kg)*DOD]
= (6341 WHrs)/[(75 WHrs/kg)*(.55)] = <u>154 kg</u>

<u>LH2/LOX Regenerative Fuel Cells</u>

The information used for sizing fuel cells comes from specifications on Apollo and Space Shuttle fuel cell systems, scaled down to meet this missions requirements (Reference 7). The Apollo fuel cells had a maximum output of 1420 watts at 70% efficiency. Studies by Rockwell International show fuel cells with 90% efficiency available in the near future if not present (the report was published in 1981 (Reference 8)). With this efficiency, the total output of the Apollo fuel cell could have been:

(1420 W)*(.90)/(.70) = 1826 watts.

Reactant consumption of the Apollo fuel cell was .55 Kg/hr, by dividing this by cell output, one gets:

(.55 Kg/hr)/(1826 W) = .301 Kg fuel mass/Kilowatt-Hour.

This can be used to determine the fuel mass required at a given output for a given period of time. The Space Shuttles fuel cells supply an average of 7 Kw each. Each unit weights 91 Kg. Specific Power is thus (7000 W)/(91 Kg) = 77 W/Kg. For the instrument bus this would warrant a 4 Kg fuel cell:

(310 W)/(77 W/Kg) = 4Kg.

Weight of the fuel for the instrument bus is:

(11.25 hrs)*(310 W)*(.301 Kg/Kw-hr) = 1.05 Kg.

Together, this equals 5.05 Kg for the fuel cell. But this kind of estimate will not account for everything, so to be conservative, the total mass of the fuel cell is estimated at 10 Kg. The fuel cell for the payload carrier is sized in a similar fashion. The fuel mass is:

(3 days)(24 hr/day)(170 W)(.301 Kg/Kw-hr) = 3.68.

Together with the cell mass of 4 Kg, this is 7.68 Kg for the fuel cell. Again, to be conservative, the total mass of the fuel cell for the payload carrier is estimated at 13 Kg.

As can be seen from the above comparison, batteries weighed quite a bit more than the fuel cells (154 kg - Ni-H2 batteries, 10 Kg LH2/LOX fuel cell). Because regenerative fuel cells are more complex than Ni-H2 batteries, they are not as reliable as batteries on long term missions but research is underway and such systems will be available around 1998. (Reference 3).

The solar array sizing for the instrument bus is as follows (Reference 1):

PT = Total Power Te = Eclipse Time = 11.25 hrs
PL = Load Power = 310 W Ts = Sun exposure Time = 13.47 hrs
Solar cell degradation = 30% over 5 years
Cr = concentration ratio = .88
S = solar constant = 600 W/m2 (at Mars)
e = cell efficiency at 25C = .205 (reference
A = active cell area
= temperature degradation factor = .005
T = cell temperature = 50 C
Array area density = 1.59 kg/m2

PT = PL + (PL*Te)/Ts
= 310 W + (310 W * 11.25 hrs)/13.47hrs
= 569 W

PT(after 5 years) = (569 W)/(1-0.3) = 813 W

A = PT /[S * Cr * e * (1 - (T-25C))]
= 813W/[(600 W)*(.88)*(.205)*(1-.005(50C-25C))]
= <u>8.6 m2</u>

Array mass = (8.6 m2)*(1.59kg/m2) = 13.7 kg
To be conservative, array mass equals <u>15 kg</u>.

For the spacecraft, the fuel cell will only need to be recharged enough to handle the small eclipse time it encounters while in its parked Martian orbit awaiting decent to the surface. The same equations apply.

PL = 170 W Te = 1.28 hrs Ts = 23.44 hrs
Solar cell degradation = .12 over 2 years
All other variables are as above.

PT = 170 W + [(170 W)*(1.28 hrs)]/(23.44 hrs)
= 179.3 W

PT(after 2 years) = (179.3 W)/(1-0.12) = 203.8 W

A = (203.8 W)/[(600 W)*(.88)*(.205)*(1-.005(50C-25C))]

= <u>2.15 m2</u>

Array mass = (2.15 m2)*(1.59kg/m2) = 3.42 kg
To be conservative, array mass equals 5 kg.

The dimensions of the fuel cells are again scaled down figures from the Space Shuttle fuel cell system. The dimensions of the Space Shuttle's 7Kw, 91Kg fuel cell are .35m x .38m x 1.01m (Reference 7). This gives a specific volume of (.35*.38*1.01 m3)/91 Kg = .0015m3/kg. The two cells weigh 10kg and 13kg. The scaled down fuel cells for this mission are estimated as follows:

Instrument Bus (10kg)*(.0015m3/kg) = .015m3 => .2m x .15m x .5m

Spacecraft (13kg)*(.0015m3/kg) = .02m3 => .2m x .2m x.5m

Finally, to integrate and control the power system a power subsystem electronics unit is responsible for: fuel cell charging, signal conditioning, power control and distribution, and power system fusing. It shouldn't weigh more than 30 kg and has estimated dimensions of .4m x .4m x .4m.

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Figure 7-1. Solar Arrays -vs- RTG Specific Power

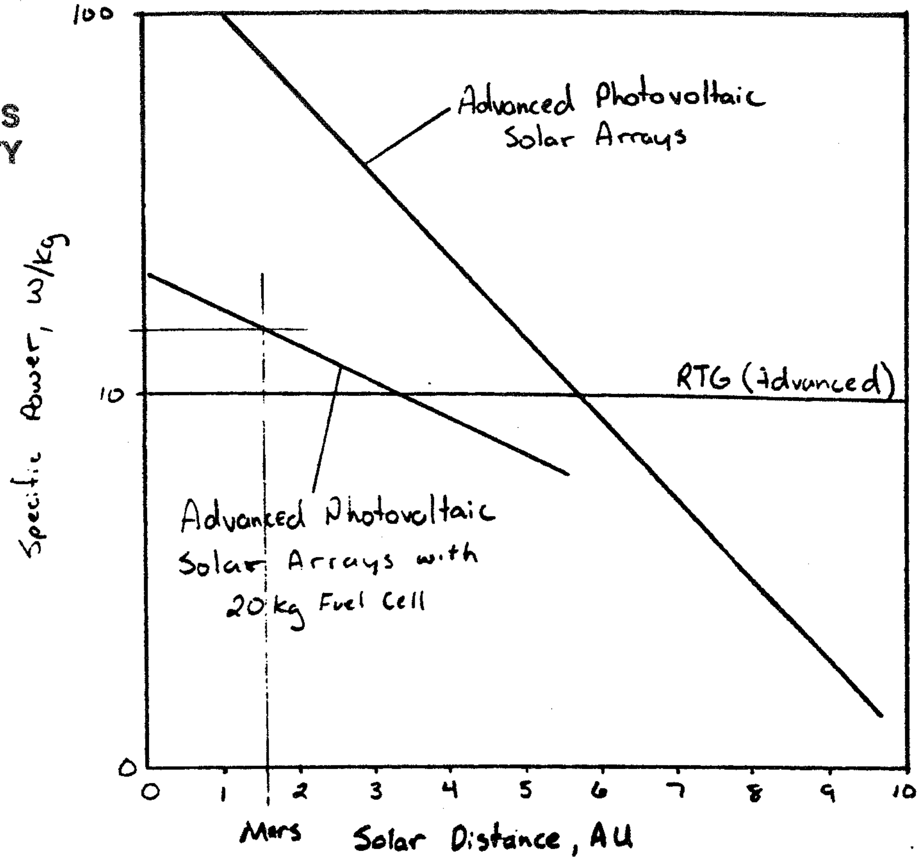
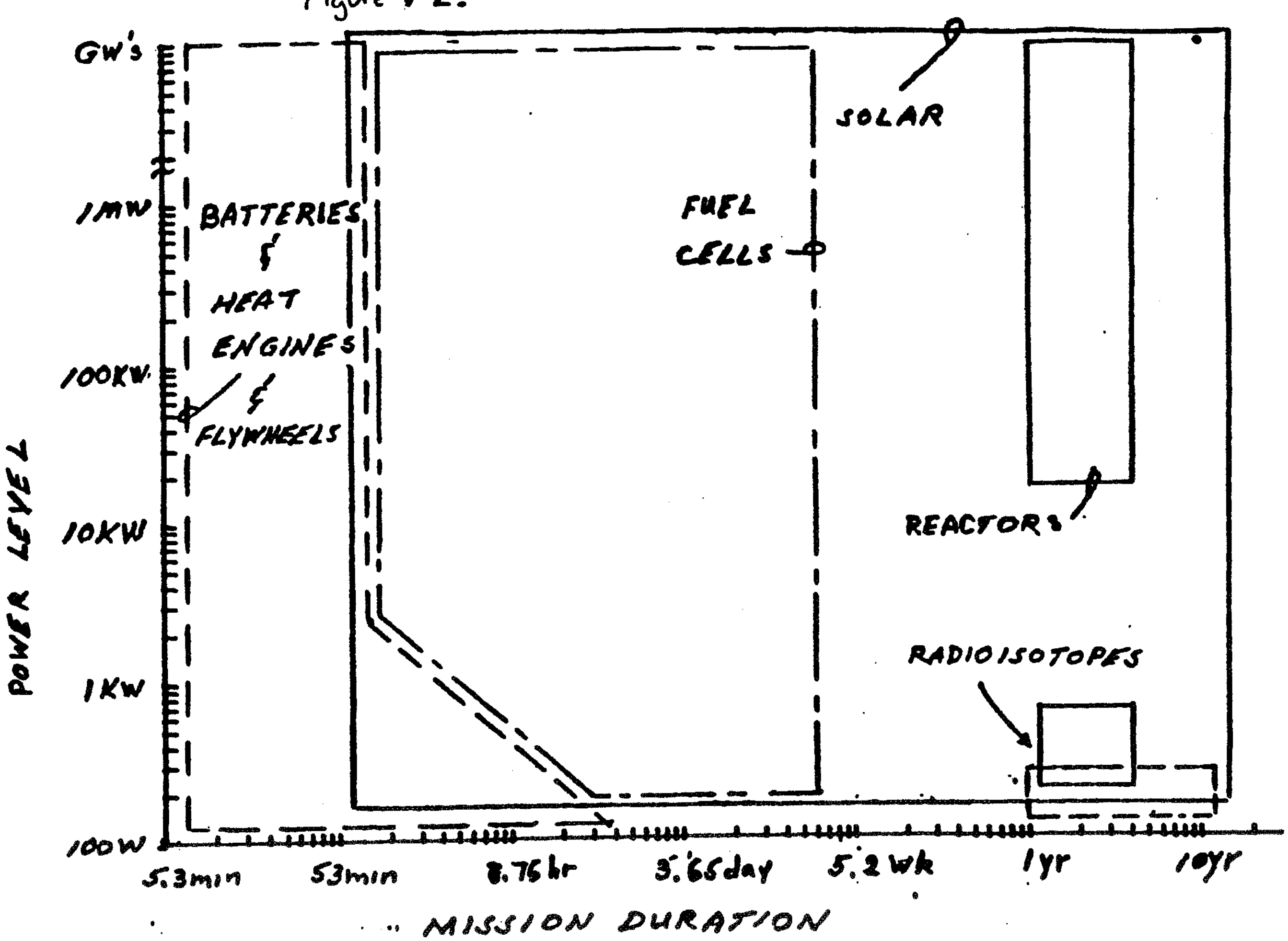


Figure 7-2. Power Source Domain



PROPULSION SYSTEM

There are seven different propulsion systems required for this mission: 1) the main propulsion system required to send the spacecraft from Earth orbit onto a Mars trajectory, 2) propulsion system required for Aerobraking, 3) instrument bus booster to put it in synchronous Martian orbit, 4) retro rockets to initiate decent to Mars surface, 5) landing rockets to slow the decent of the lander to the surface of Mars and 6) two AAC Thruster systems (one for the spacecraft and one for the instrument bus). This report will deal primarily with the main propulsion system and the instrument bus booster system. General requirements for the other systems will be looked at but most of those calculations will be covered in the other subsystem reports (propulsion system for Aerobraking, retro rockets and landing rockets are covered in the Aerobraking subsystem report and the AAC Thruster systems are covered by the AACS subsystem report.)

For the main engine, two propulsion systems were seriously considered, chemical propulsion and ion propulsion. Chemical propulsion has been NASA's work horse since the start of the space race. It is a highly reliable propulsion system and is relatively cheap (research costs have already been payed for and most of the components of a chemical rocket engine can be pulled off a shelf). Ion propulsion on the other hand is still in its research phase. Since it is new technology, costs will be high and its reliability can only be speculated. The nice advantage of ion propulsion is its high Isp, an MPD Ion propulsion system has an Isp equal to about 6000 s (compared to a chemical rocket's Isp of around 460 s for liquid hydrogen/liquid oxygen (LH2/LOX)) (Reference 1). This high Isp means a large decrease in system weight (when comparing to low Isp propulsion systems) as can be tested using the rocket equation (delta V = g*Isp*ln(Mi/Mf), g = 9.8m/s, Mi = initial mass, Mf = final mass) (Reference 10). But, the major problem area with ion propulsion is its low thrust which means a longer mission profile (to give the engine the time to attain delta V). In a lecture by John K. Soldner of SAIC called "Pathways To Mars" (Reference 11), he illustrated the mission time difference between a ballistic trajectory (chemical propulsion) verses a low thrust trajectory (ion propulsion). For a mission to arrive at Mars in August 2008 with chemical propulsion it took an Earth launch date of September 2007 for a total of 11 months. For the ion propulsion system, it took a launch date of September 2006 to reach Mars by August 2008, a total of 23 months. It was felt by this design team that this extra year of flight time would put a burden on all the other spacecraft subsystems in terms of increasing reliability requirements and the extra cost involved with ground monitoring and system management outweighed the higher spacecraft weight disadvantages of chemical propulsion.

The main propulsion system will be a LH2/LOX pump feed chemical rocket. The same type used by the upper stages of the Apollo missions and the type used for the main engines of the Space Shuttle. A LH2/LOX system was chosen because of its high Isp, the latest designs are capable of delivering an Isp of 470 s (Reference 2). As mentioned before, the higher the Isp the lower the initial mass (fuel mass primarily) will be. A disadvantage of the LH2/LOX system is it requires refrigeration to keep the LH2 and LOX tank below their boiling point which in turn requires extra mass an power. (note: The power to maintain refrigeration will be delivered from a separate power system built into the main engine stage. Further studies are necessary to determine the power required.) A quick trade study between monomethylhydrazine/nitrogen tetroxide (MMH/NTO)(a fuel with one of the highest Isp's (340s) for a system that doesn't require refrigeration(Reference 1)) and LH2/LOX puts the advantage of the LH2/LOX system into perspective:

MMH/NTO 3569m/s=(9.8m/s)*(340s)*ln(Mi/9000) Rocket Eqn.

Mi = 26268 Kg

LH2/LOX 3569m/s=(9.8m/s)*(470s)*ln(Mi/9000) Rocket Eqn.

Mi = 19533 Kg

A difference of 6735 kg which is far more than any refrigerant system would cost in weight for this small a propulsion system.

The actual fuel mass and engine and tank calculations for this spacecraft are as follows:

delta V = 3569m/s
Mt = total mass
Msc = spacecraft mass = Mt-(Mf+Met) = 7550 kg
Mf = total mass of the fuel
Met = total mase of the engine & tanks = 15% of Mf
g = 9.8m/s
Isp = 470s (LH2/LOX)

delta V = g * Isp * ln[Mt/(Msc+Met)]

3569m/s = (9.8m/s)*(470s)*ln[Mt/(7550+Met)]

By manipulating this equation,

Mt = 20200 kg

$\underline{Mf = 11000\ kg}$ (with a 300 kg fuel contingency mass)
$\underline{Met = 1650\ kg}$

LOX/LH2 ratio = 6
LOX density = 1141 kg/m3
LH2 density = 70.79 kg/m3

LOX mass = (6/7)*(11000kg) = 9429 kg
LH2 mass = (1/7)*(11000kg) = 1571 kg

LOX volume = (9429kg)/(1141kg/m3) = 8.26 m3
LH2 volume = (1571kg)/(70.79kg/m3) = 22.19 m3

There will be three tanks total on the main engine stage one LOX tank and two LH2 tanks.

volume of a sphere = (4/3) * pi * r ** 3

LOX tank 8.26 m3 = 4.189 * r **3

$\underline{r = 1.254\ m}$

LH2 tank (22.19 m3)/2 = 4.189 * r ** 3

$\underline{r = 1.384\ m}$ (2 tanks)

The engine's estimated size is 2.5m x 1.0m x 1.5m.

(Note: The engine and tank mass are taken to be 15% of the fuel mass. This a simple estimate mentioned in reference 4 to give good and quick ball park figures to be used in initial design.)

The LH2/LOX stage (engine and tanks) will be jettisoned as soon as the delta V burn is over to alleviate the spacecraft of unnecessary mass.

The LH2/LOX system mentioned above was possible because it was used at the very start of the mission and the propellant did not have to be refrigerated for a long period of time. But, for the rest of this missions propulsion systems this is not true. All of the other systems will need propellants which are storable at least up to a year, for the instrument bus it will be 4 years. The primary concern for these remaining systems is that they be reliable and that they add a minimal amount of mass to the spacecraft. Two systems were looked at 1) monopropellant hydrazine and 2) bipropellant MMH/NTO. Monopropellant hydrazine is less complex, just one fuel tank and one set of fuel lines is required for this system. But the bipropellant has much better performance (Isp for MMH/NTO = 340, Isp for hydrazine = 220 (Reference 1)) with only a slight gain in complexity due to multiple fuel feed systems. Both have been tested extensively. In fact, bipropellant MMH/NTO was used on the Viking Program to Mars in 1975-76 (Reference 6). In terms of weight the same technique as shown above can be used to show that because of the higher Isp the MMH/NTO system will have less total mass than the hydrazine system would. For these reasons, MMH/NTO systems have been chosen as the propulsion system for all the remaining propulsion requirements of this mission.

The instrument bus will need an MMH/NTO propulsion system to park it in synchronous Martian orbit. The fuel mass and engine and tank calculations are as follows:

delta V = 1563.1 m/s
Ms = satellite mass = 330 kg
g = 9.8m/s
Isp = 340s

1563.1 m/s = (9.8m/s)*(340s)*ln(Mt/(330kg+Met))

<u>Fuel mass = 700kg</u> <u>Engine & tank mass = 105kg</u>

To be conservative, the fuel mass is actually a little more than necessary (delta V calculated at the above values = 1585m/s).

NTO/MMH mass ratio = 1.6
NTO density = 1431 kg/m3
MMH density = 870.1 kg/m3

MMH mass = 700kg/2.6 = <u>269.2 kg</u>
NTO mass = 700kg - 269.2 kg = <u>430.8 kg</u>

MMH volume = 269.2kg/(870.1kg/m3) = .309 m3
NTO volume = 430.8kg/(1431kg/m3) = .301 m3

A total of six tank will be used to house the above fuel (3-MMH, 3-NTO) due to space limitations. The radius of the tanks are as follows:

volume of a sphere = (4/3) * pi * r ** 3

MMH tanks (.309 m3)/3 = 4.189 * r ** 3

<u>r = .29 m (3 tanks)</u>

NTO tanks (.301 m3)/3 = 4.189 * r ** 3

<u>r = .29 m (3 tanks)</u>

The MMH/NTO systems used on this mission are small enough that they will be pressure feed rather than pump feed. This means pressure tanks must be sized as follows (Reference 10).

Gas used: Helium

Pp = estimated propellant tank pressure = 1696000 Pa
Vp = propellant volume = .309 m3 + .301 m3 = .61 m3
R = 230 J/kg K
To = tank temperature = 290 K
k = 1.667 (reference 9)
Pg = 1.696 Mpa
Po = estimated helium tank pressure = 20 Mpa
density of liquid helium = 141.6 Kg/m3

Mass of helium = (Pp * Vp * k)/(R * To * (1-Pg/Po))
= <u>28.25 kg</u>

Volume of helium = (28.25kg)/(141.6 kg/m3)
= .2 m3

<u>radius = .362 m</u>

The propulsion systems looked at in the Aerobraking and AACS subsystem sections were sized in the above manner.

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A.C.M.E

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COMMAND, CONTROL, AND COMMUNICATIONS

SCIENCE

MISSION MANAGEMENT, PLANNING & COSTING

John C. Mutka

[redacted]

ASSIGNMENT

The project objective is to develop a conceptual design for the spacecraft system required to deliver a manned aircraft to the Martian surface.

SYSTEM REQUIREMENTS

1) Launch/Delivery of Spacecraft/Aircraft shall occur during the time period of 2005-2010.

2) Spacecraft (S/C) will consist of two primary components:

   - Instrument Bus
   - Payload Re-entry System

3) Instrument Bus will remain in orbit after seperation from Payload Re-entry System.

4) The following subsystems will be identified for facilitating system integration:

   - Aerobrake
   - Attitude & Articulation Control
   - Command & Data Control
   - Mission Management, Planning & Costing
   - Power & Propulsion
   - Science & Radio Relay Instrumentation
   - Structure

5) Stress simplicity, reliability, minimum mass & low cost.

6) Achieve Mission Science Objectives as outlined in document "AAE 241 Mission Science Objectives."

7) Four S/C will be built: three flight ready, and one for testing.

8) S/C and Aircraft (A/C) will be delivered to a low Earth orbit in a Space Shuttle (STS), and assembled at a Space Station.

9) S/C will be retrievable by a remote manipulation device on the STS or Space Station.

10) Off-the-shelf hardware should be used where possible.

11) Should not use materials or techniques expected to be available after 1998.

12) Artificial Intelligence (AI) will be used where applicable.

13) Lifetime of S/C should be at least four years, with nothing designed to limit even longer lifetime.

14) Nothing in S/C design will preclude it from performing several different types of missions.

MISSION PLANNING SUBSYSTEM REQUIREMENTS

1) Define target locations necessary for the S/C to fulfill the System Requirements.

2) Analyze pathways between locations, attempting to minimize ΔV.

3) Design, by working with and integrating between the other six subsystems, a S/C to fulfill Requirements one and two above (Design should include testing, launch, and mission support necessary for the S/C).

4) Estimate cost for S/C design, according with System Requirements (Attempting to minimize such costs).

MISSION ANALYSIS

(Launch, Trajectory and Orbit)

LAUNCH

Delivery of S/C and A/C to the Space Station

- S/C and A/C will be modularized and packaged to be compatible with the STS cargo bay. (See Structures)
- It will take three STS trips to completely transport the S/C and A/C to the Space Station.
- The fuel for propulsion will be supplied at the Space Station. (See Propulsion)
- An orbit of 250 miles (400 km) will be used for the Space Station.
  (This is an arbitrary choice.)
  [Note: ΔV orbit departure decreases 0.009 km/sec when moving from 250 mi. (400 km) to 500 mi. (805 km), which is considered negligible for this analysis.]
- Time to complete this stage is dependent upon the STS system.

S/C and A/C at the Space Station

- Assembly and fueling of modularized S/C and A/C. (See Structures and Propulsion)
- Ability to retrieve and/or move about by remote manipulation device aboard the STS or Space Station. (See Structures)
- Pre-launch analysis and testing of S/C components. (See each subsystems)
- Space Station is advanced, so there are no immediate dangers and/or problems with launching a high/low thrust S/C from the Station.
- Time to complete this stage is approximately one week.

TRAJECTORY

Method of Attack and Results

- Two feasible pathways were determined:
  a) Earth - Mars (EM)
  b) Earth - Venus Flyby - Mars (EVM)
- Five maneuvers during flight were also defined:
  a) Impulsive firing of engines (IMP)
  b) Jettison stage mass after firing (MASS)
  c) Aerobraking (AERO)
  d) Gravity assist by flyby of Venus (GRAV)
  e) Constant firing of engines (CONB) (i.e., ion propulsion)
- Four factors of greatest importance to trajectory choice were defined and given a total pt. value:
  a) Total ΔV of pathway (45 pts.)
  b) Simplicity of pathway (35 pts.)
  c) Flight time of pathway (25 pts.)
  d) Ability to solve for above (20 pts.)
- The following Table I was produced, where total # of pts. possible = 125. (125 excellent, 0 unsatisfactory)
- From Table I, five of the higher pt. total trajectories were then further studied:
  a) IMP - IMP (EM)
  b) IMP - MASS - AERO (EM)
  c) IMP - AERO (EM)
  d) IMP - IMP - IMP (EVM)
  e) IMP - IMP - AERO (EVM)

  with a final Mars orbit = 250 mi. (400 km). (See Orbit)

  a) IMP - IMP (EM)

  Found ΔV_{min} total mission for window of 2005-2010 in 100 day increments, flight times of 150-500 days in 50 day increments. Results shown on Graph I.

  This determined basic cycle of optimum ΔV's.

Examined "valleys" of window in 3, 180 day blocks, 20 day increments for flight times 150-500 days. Results shown on Graphs II, III, & IV.

<u>ΔV_{min} = 5.75 km/sec. & Flight time = 350 days</u>

b) IMP - MASS - AERO (EM)

Refine data from part a), giving ΔV Earth orbit departure and V_{∞} for Mars arrival. Results shown on Graphs II, III, & IV. On Graph V, further examination of the lowest "valley" is shown. (18 day window, in 2 day increments for flight times of 310-345 days, in 5 day increments)

Giving ΔV departure to Propulsion, jettisoned mass is determined. Mass, along w/ V_{∞} is given to Aero. Using an 11 day launch window, the ΔV_{min} depart. is specified to be 3.65 km/s, and V_{∞} is 2.5 km/s.

<u>ΔV_{min} = 4.41 km/sec. & Flight time = 374 days</u>

c) IMP - AERO (EM)

Follow procedure and results of b), except in Aero. where, because of no mass jettisoned, will have larger mass to place in orbit. This increases flight time and ΔV_{min} of Aerobraking.

d) IMP - IMP - IMP (EVM)

Found ΔV_{min} for total mission, for launch window of 2005-2010, in 50 day increments, optimizing flight time. Then examined the only true "valley", in a 45 day block, 5 day steps for opt. flight time. Results are shown top of pg. on Graph VI.

<u>ΔV_{min} = 8.74 km/sec. & Flight time = 471 days</u>

e) IMP - IMP - AERO (EVM)

Using Graph VI, the procedure is the same as part b), except no mass loss. As shown, the ΔV_{min} is greater to arrive at Mars and V_{∞} increases as well. These lead to longer flight times & greater ΔV_{min}.

- Summary of Trajectories Studied:

TRAJECTORY	ΔV_{min} Total [km/s]	FLIGHT TIME [days]
a) IMP - IMP (EM)	5.75	350
b) IMP - MASS - AERO (EM)	4.41	374
c) IMP - AERO (EM)	> 4.41	> 374
d) IMP - IMP - IMP (EVM)	8.74	471
e) IMP - IMP - AERO (EVM)	> 4.41	> 471

[redacted]

- TRAJECTORY TO BE USED ..... IMP - MASS - AERO (EM)
- LAUNCH WINDOW ..... OCT. 8, 2009 thru OCT. 19, 2010

[redacted]

Further analysis of trajectory

- Location of Earth and Mars during mission sequence:
  See Chart I

  E_D = position at departure 10/(8-19)/09

  E_A = position at Mars arrival (8-9)/(24-5)/10

  E_O = position after aerobraking (10-11)/(20-1)/10

  E_{LP} = position at end of transfer orbit about Mars 11/(17-28)/10

  Also have M_D, M_A, M_O & M_{LP}.

TABLE I: TRAJECTORY ANALYSIS

EM Pathway	Total ΔV [km/s]	Simple	Flight Time [days]	Solve	Total
1) IMP - IMP	21	35	25	20	101
2) IMP - MASS - IMP	23	33	25	18	99
3) IMP - AERO	34	32	21	17	104
4) IMP - MASS - AERO	40	30	21	16	107
5) MULT. IMP	24	25	23	15	87
6) MULT. IMP - MULT. MASS	26	23	23	14	86
7) MULT. IMP - AERO	37	22	19	13	91
8) MULT. IMP - MULT. MASS - AERO	43	20	19	11	93
9) CONB) see	44	5	5	-10	44
10) CONB - AERO) Matt Zell	45	2	1	-10	38
EVM Pathway					
11) IMP - IMP - IMP	19	31	22	19	92
12) IMP - GRAV - IMP	21	27	19	16	83
13) IMP - IMP - AERO	32	28	20	17	97
14) IMP - GRAV - AERO	34	24	17	14	89
15) MULT. IMP	22	21	20	14	77
16) MULT. IMP - AERO	35	18	18	12	83
17) MULT. IMP - GRAV	24	14	17	11	66
18) MULT. IMP - GRAV - AERO	37	11	15	9	72
19) CONB) see	42	3	3	-12	36
20) CONB - AERO) Matt Zell	43	0	-1	-12	30

[Note: MASS could also be included in the possiblities of EVM. However, due to little increase in total pts., it has been omitted for space.]

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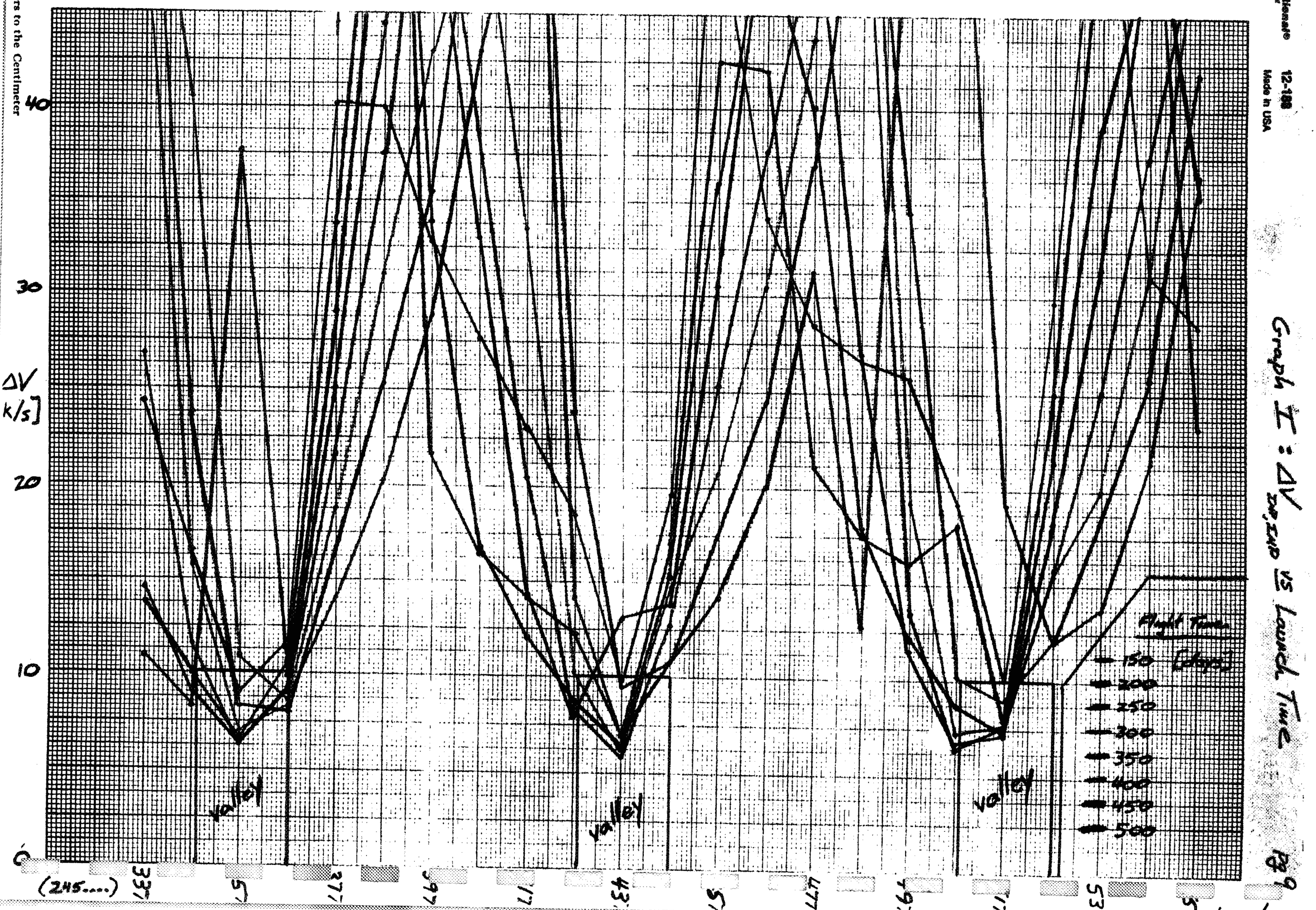

10 Millimeters to the Centimeter
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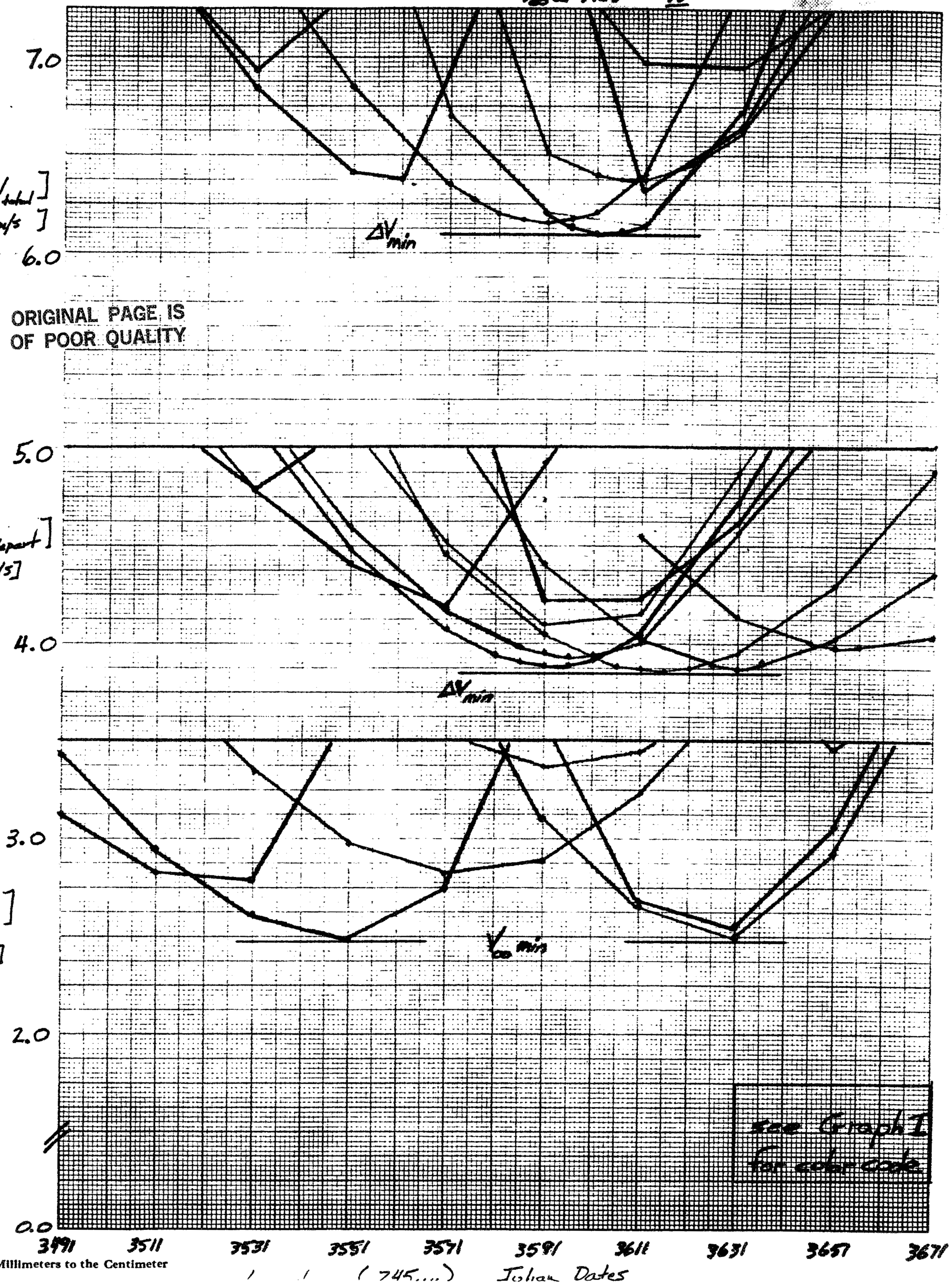
Graph II : ΔV_{total} vs / $\Delta V_{total\ departure}$ vs / V_∞ at Mars vs — Time of Launch



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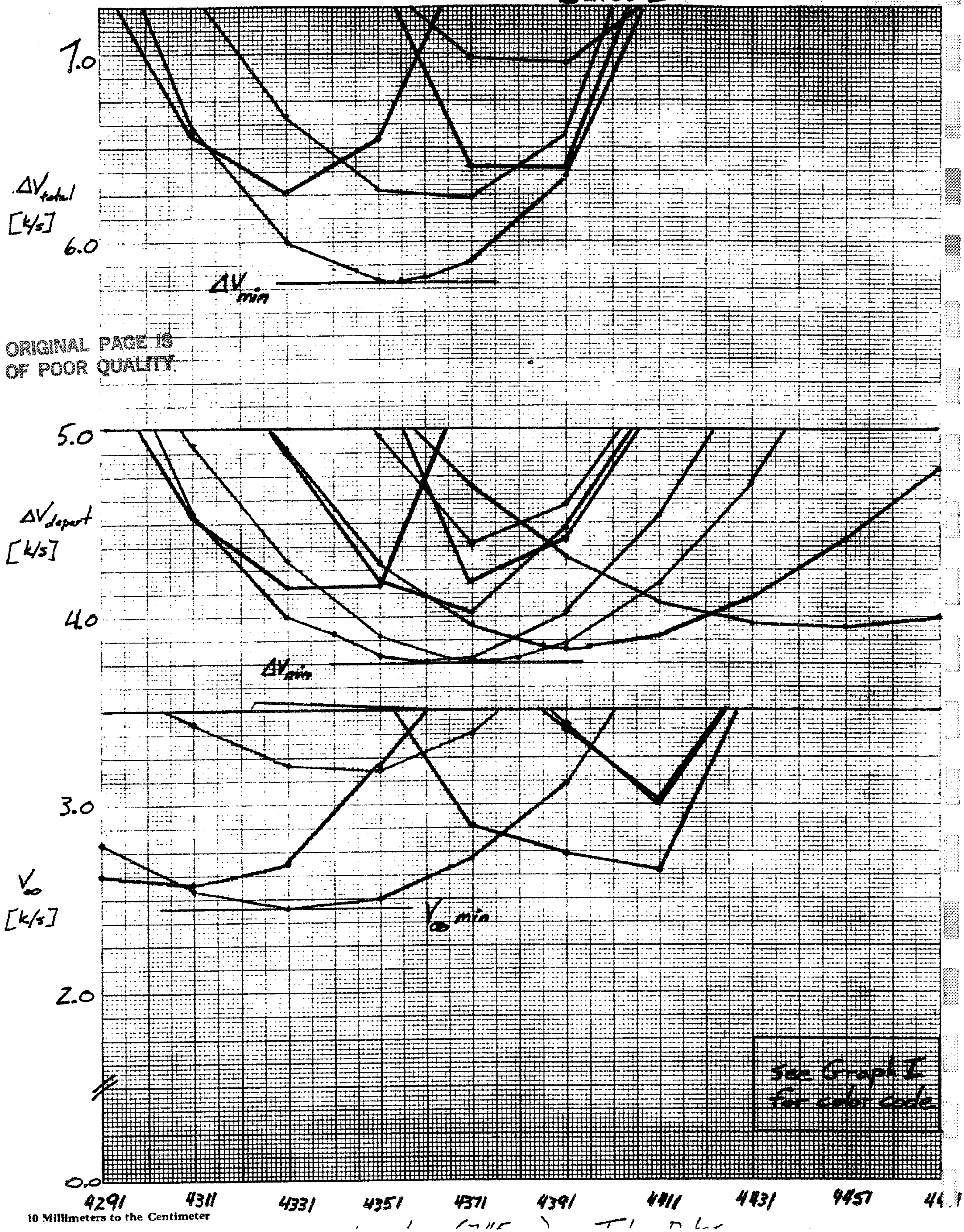
Graph III : ΔV total vs
ΔV depart vs
V∞ at Mars vs Launch Time



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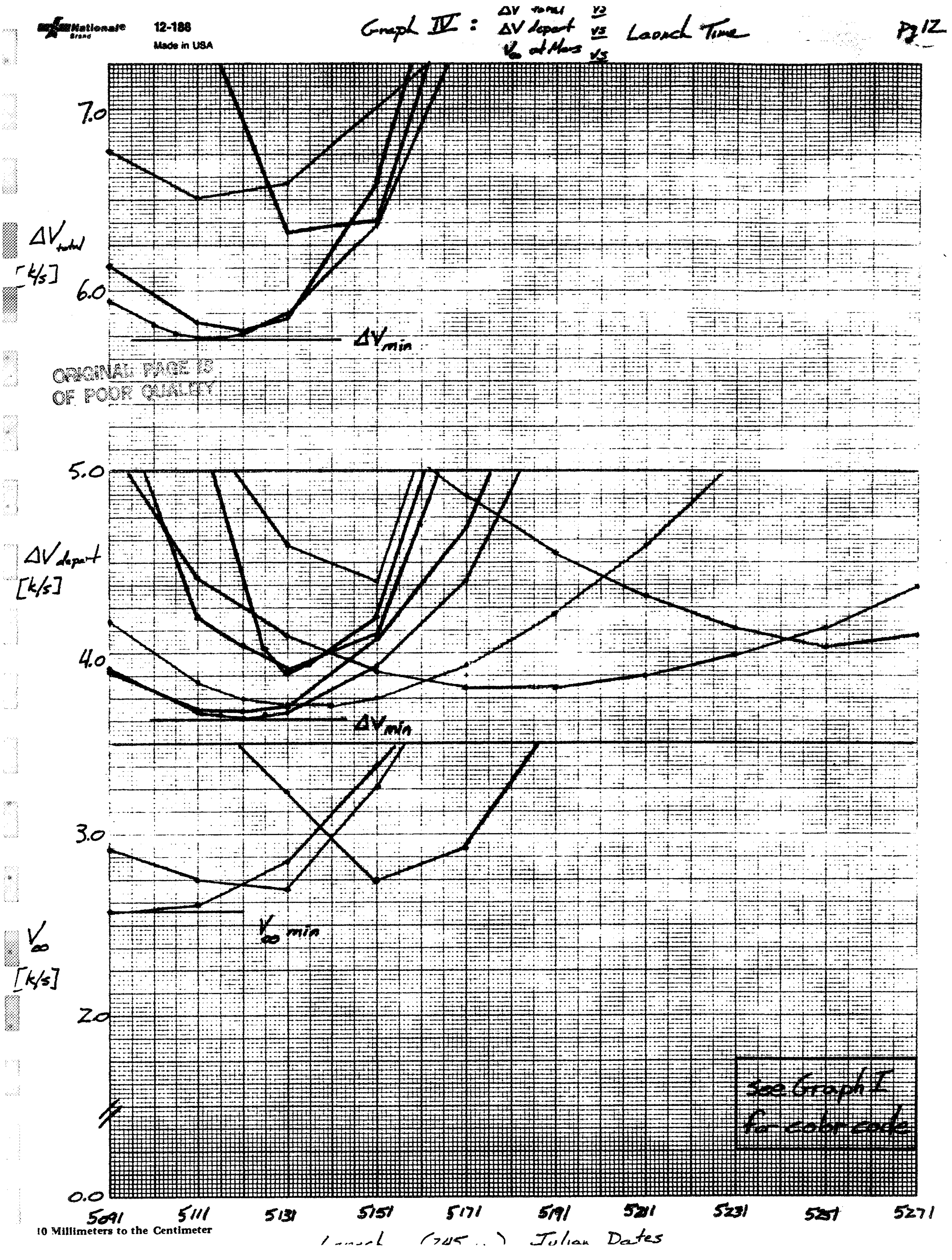

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Graph IV : ΔV total vs ΔV depart vs V∞ at Mars vs Launch Time
Pg 12
7.0
ΔV total [k/s]
6.0
ΔV min
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5.0
ΔV depart [k/s]
4.0
ΔV min
3.0
V∞ min
V∞ [k/s]
2.0
See Graph I for color code
0.0
5091
5111
5131
5151
5171
5191
5211
5231
5251
5271
10 Millimeters to the Centimeter
Launch (245..) Julian Dates

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Graph V : ΔV_{depart} vs V_∞ at Mars vs Launch Time

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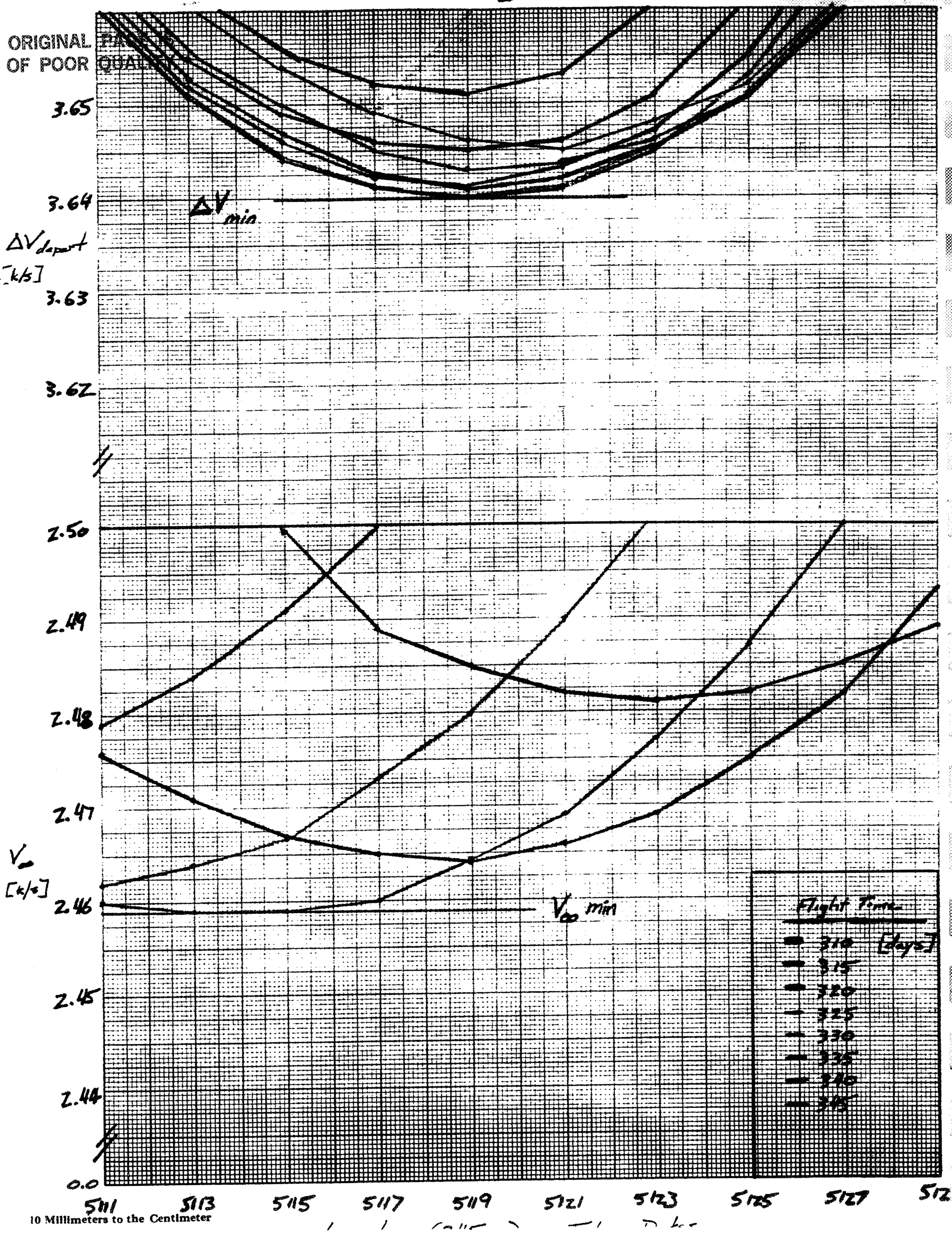

10 Millimeters to the Centimeter

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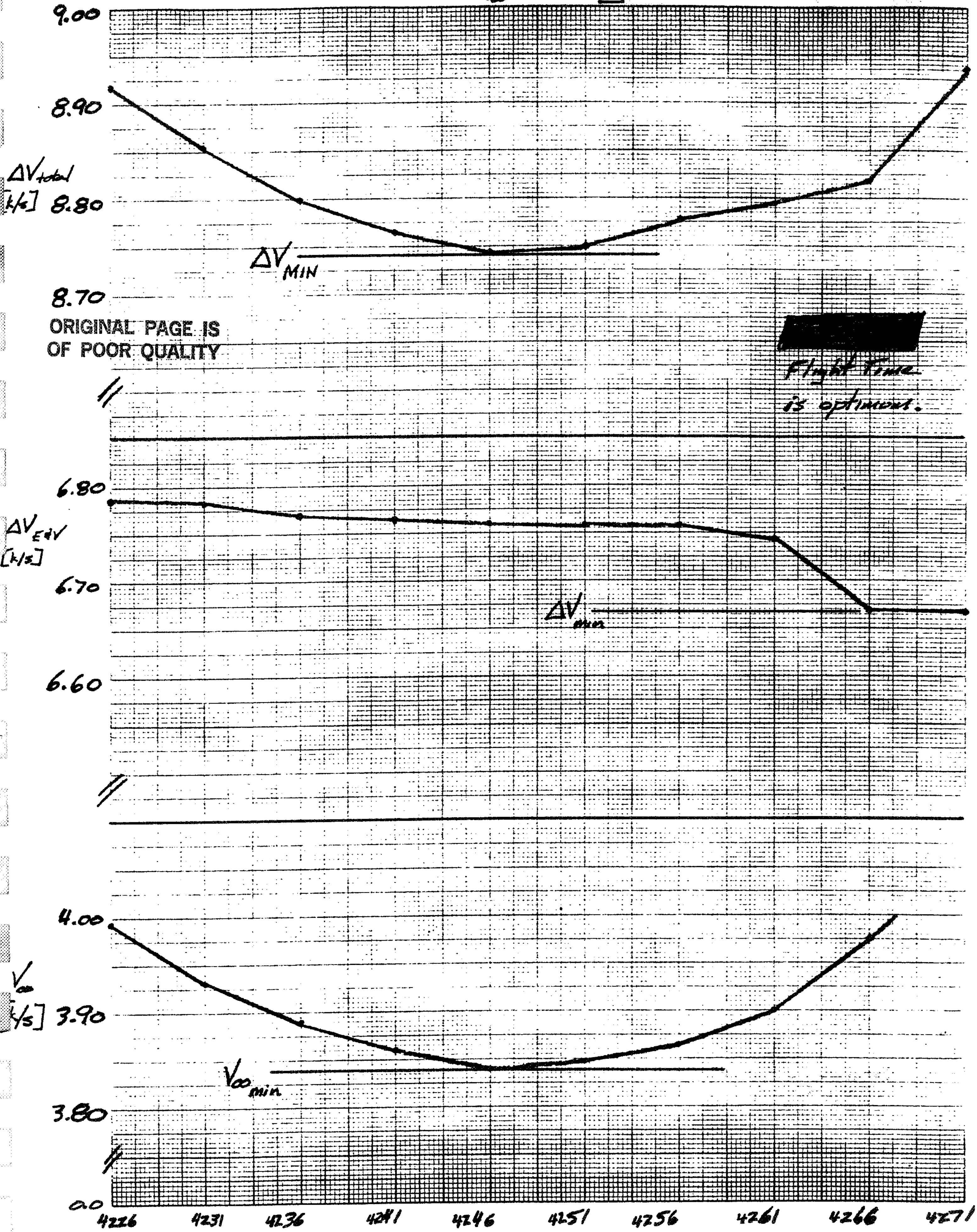
Graph VI: $\Delta V_{EARTH + Flyby}$ vs ΔV_{Total} vs V_∞ at Mars vs Launch Time



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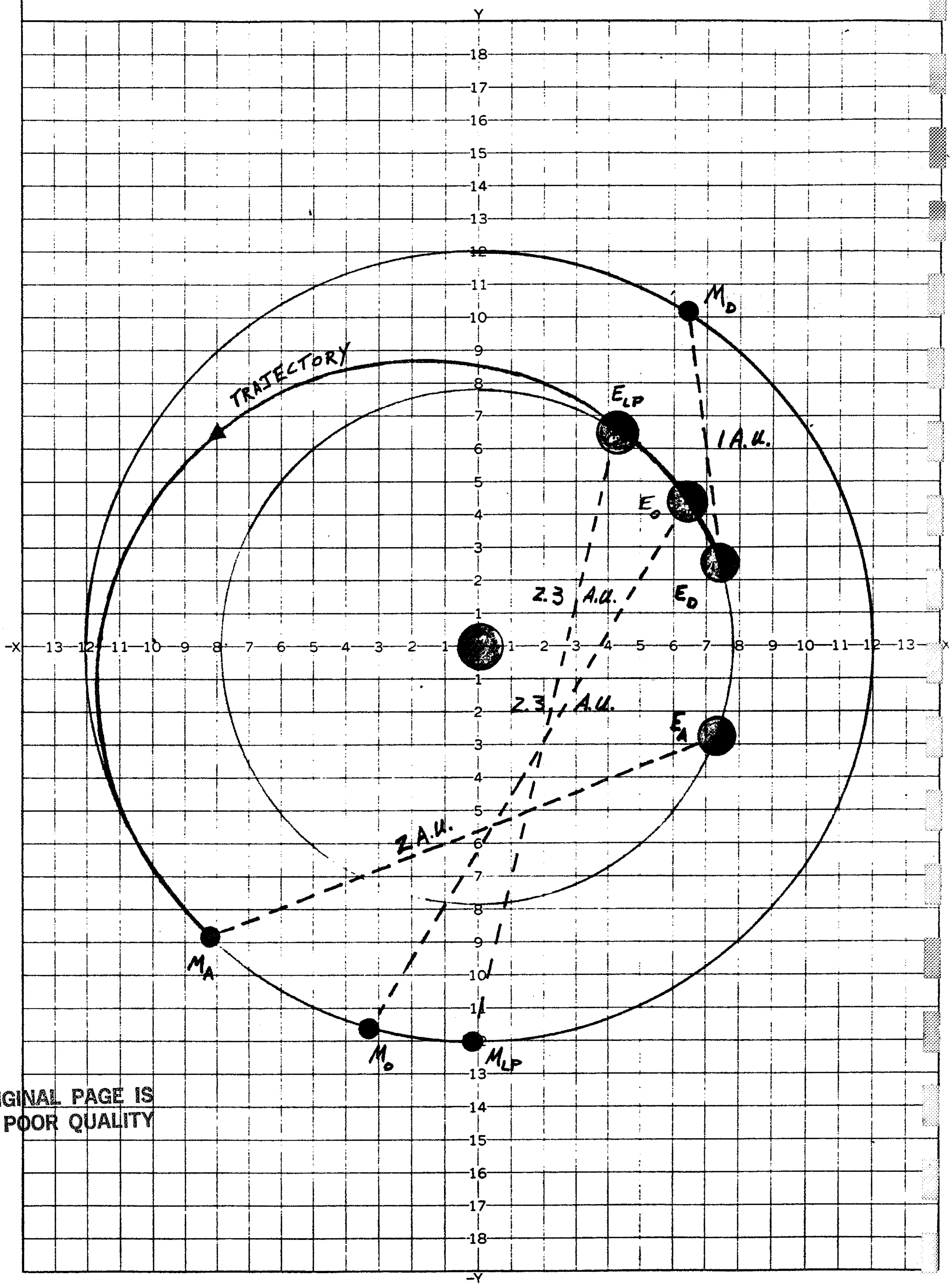
10 Millimeters to the Centimeter

Launch (245) Julian Dates

Chart I : Locations of Earth & Mars



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Quadrant Coordinates

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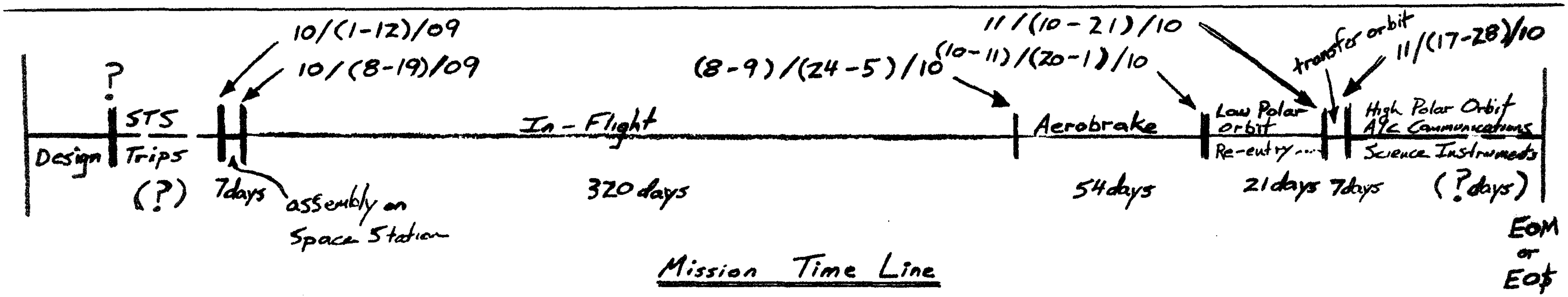
Table II: Evaluation of possible orbits about Mars for S/C

Orbit	Determine Land Site Suitability	Minimize Energy/Time Re-entry	Observ. Total Planet	Night & Day Observ.	Communicate w/Base & A/C	St[illegible] of Orbit	[illegible] from [illegible] w/ Moons	Total Re-entry	Total Instr. Bus	Total
[illegible]										
Polar	6	7	10	10	2	8	10	13	22	**53**
[illegible]	6	7	0	10	2	8	0	13	0	31
Equator	10	10	0	10	4	8	10	20	0	38
Middle										
Polar	4	5	9	8	5	9	9	4	22	49
[illegible]	4	5	0	8	5	9	9	4	0	27
Equator	8	8	0	8	6	9	9	16	0	34
Elliptical										
Polar	3	4	8	5	3	7	5	7	16	35
[illegible]	3	4	3	5	3	7	3	7	11	28
Equator	6	5	1	0	5	7	2	11	11	31
(Synchronous)										
Polar	2	3	8	6	**7**	10	10	5	21	**[illegible]6**
[illegible]	2	3	5	6	7	10	10	5	18	43
Equator	4	4	0	6	10	10	10	8	0	28
Heliocentric	1	1	3	0	8	1	10	2	0	13

Scale: 1 (worst) – 10 (best) [0 indicates failure, orbit is eliminated from consideration for category]

Conclusion of Evaluation: Maintain Low Polar orbit out of Aerobrake. Detach Re-entry system (See Aerobrake) and collect science data (See Science). Allow four weeks for A/C assembly & testing, by which time, move to a high polar orbit w/ period equal to Mars day. (See Communications) Change in ΔV depart Mars from low orbit (4002 km / 250 miles) to high orbit (10403 miles / 16 742 km) is ΔV required approximately to change orbits which



Mission Time Line

Labor and Cost Analysis

[Labor Hours to Labor Costs : $20.00/hr (Cheap labor in 2009)]

Development Project (1 S/C)

Category	Mass [kg]	DLH [khrs]	RLH [khrs]	NRLH [khrs]	COST $M	Inheritance (%) BB	ER	MIN	MAJ	NEW	DLHI [Khrs]	NRLHI [khrs]	COSTI $M
Structures / Thermal Control	813.4 / 540.5	697.9 / 1913.0	205.4 / 171.4	492.5 / 1741.6	24.6 / 87.0	100 / 77	0 / 0	0 / 23	0 / 0	0 / 0	205.4 / 471.8	0 / 300.4	0 / 15.0
Propulsion	3421.0	1667.1	1529.8	137.3	6.9	53	0	47	0	0	1578.2	48.4	2.4
Attitude and Articulation	86.8	537.7	167.7	370.0	18.5	75	0	20	5	0	240.8	73.1	3.7
Telecommunications	200.0	1786.3	882.9	903.4	45.2	100	0	0	0	0	882.9	0	0
Antennas	20.0	182.5	66.8	115.7	5.8	0	25	75	0	0	137.7	70.9	3.5
Command and Data	70.0	383.9	118.6	265.3	13.3	0	0	0	100	0	370.6	252.0	12.6
Solar / Battery Power	387.0	1788.0	1310.0	478.0	23.9	82	0	18	0	0	1374.5	64.5	3.2
Aerobrake / Re-entry	500.0	650.4	104.2	546.2	27.3	11	0	0	89	0	513.9	409.7	20.5
Altimeter	7.0	73.6	13.6	60.0	3.0	0	0	100	0	0	58.6	45.0	2.3
Remote Sensing Instr.	64.5	314.8	26.1	288.7	14.4	6	0	56	0	38	257	230.9	11.5
Total Hardware →	6110.2	9995.1	4596.5	5398.7	269.9	—	—	—	—	—	6091.4	1494.9	74.7
Total Cost Reduction →													72.3%

★ 60% of S/C considered BB!

	DLH	COST $M	DLHI	COSTI $M
System Support	3049	152.5	1875.3	93.8
Launch + 30days	980.3	49.0	597.4	29.9
Program Management	1041.7	52.0	650.0	32.5
Total Development	15066.1	753.3	9214.1	460.7
Flight Project (1 S/C)				
Mission Operations MD=48 ED=15	1854.3	927.2	1377.7	68.9
Data Analysis	788.1	39.4	585.5	29.3
[illegible] Program	17[illegible]8.5	88[illegible].4	1[illegible]7.3	[illegible]9.9

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JCM-17

POWER AND PROPULSION

Pamela Warmack

[redacted]

SUBSYSTEM REQUIREMENTS

Many of the requirements that the Power and Propulsion Subsystem (PPS) tried to meet are included in the system requirements listed in the Mission Planning (MP) section. These include requirements such as stress on simplicity, reliability, low mass, low cost, and off-the-shelf hardware wherever possible. The PPS specific requirements are as follows:

-- Provide thrust necessary for successful completion of mission. Required thrust includes delivery vehicle, instrument bus, payload retro-rockets, and all Attitude and Articulation Control System (AACS) thrusters.

-- Provide uninterrupted power to spacecraft loads during mission life.

-- Protect power bus and units against damage due to load faults.

-- Protect user loads against outages and damage due to power unit failures.

-- Control and process power source and energy storage device outputs into forms compatible with other subsystem and payload needs.

-- Send telemetry to and accept commands from Command, Communications and Control System (CCC).

-- PPS must be self-powered.

METHOD OF ATTACK

The method of attack used in this design was fairly simple. The first step was distilling general and subsystem-specific requirements from the Request for Proposal (RFP). Next, a literature search was run to find power and propulsion system (PPS) options. These options were analyzed and compared by how well each option would meet the RFP requirements, stressing simplicity, low cost and mass, and reliable technology which would be available by 1998. An intensive literature search was run on the most feasible possibilities, so further analysis could be done to make final PPS selections. Information was gathered from other subsystems to allow calculation of PPS sizing. If components proved to be of reasonable dimensions, then the systems were integrated into the spacecraft. If the sizes were not feasible, more research was done in an attempt to modify the system. Otherwise, a new system would be chosen.

PROPULSION

LOW THRUST VS. HIGH THRUST

The first decision to be made about the propulsion system was to choose between a low and a high thrust system. High thrust propulsion seemed to be the obvious selection based on a number of reasons. Several low thrust options are at a relatively low level of development. The most mature of the low thrust systems, the ion engine, has several major disadvantages associated with it. Currently, sufficient resources are not available to allow Mission Planning to explore various trajectories or to calculate these trajectories accurately. The cost of developing a low thrust trajectory software package comparable to MULIMP, used for high thrust trajectories, would be prohibitive at this point.

While ion engines have a high specific impulse (I_{sp}) of greater than 2000 seconds, the resultant thrust is so small that the time required to complete a mission is greatly increased.[4,5,6] Some may argue that time is not critical, since the delivery spacecraft is not manned, however the overhead costs of ground support for the extended duration must be taken into account. Also, the assumption was made that the Mars base would need the exploratory aircraft as soon as possible so the necessary data could be gathered to plan future Mars missions.

Finally, while ion engines have been built and tested, there is some doubt that this technology would have mission-grade reliability by the 1998 restriction date, especially for a mission of this importance. When all these factors are considered, a high thrust propulsion system is the most viable option at this date.

CHEMICAL VS. NUCLEAR PROPULSION

Chemical and nuclear are the two most feasible kinds of high thrust propulsion systems for this mission. After careful evaluation of the two, it was determined that the originality and high I_{sp} of a nuclear propulsion system did not outweigh the high mass penalties, high costs, questionable reliability, and considerable safety and political concerns associated with it.

Most chemical systems are relatively simple and very reliable. The mass of the average chemical propulsion system is moderate, but much lower than that of a nuclear system. The safety of chemical fuel is of concern only to the point of following standard handling and storage proceedures for the particular type of propellant. Chemical propulsion systems were chosen for all propulsive functions: Earth orbit escape, aerobrake orbit insertion, instrument bus propulsion, payload retro-rocket firing, and all AACS maneuvering.

CHOICE OF PROPELLANT

Solid chemical propellants were discarded as being too inflexible in case of changing mission requirements or emergency situations. Several different liquid chemical fuels were considered. A liquid oxygen/liquid hydrogen (LOX/H) combination was discarded due to increased mass for high pressure tanks and an extensive cryogenic cooling system. Another problem with LOX/H would be the significant amount of fuel dissipation over the duration of the mission.[15]

The final selection was the bi-propellant using monomethyl hydrazine (MMH) for the fuel and nitrogen tetroxide (NTO) for the oxidizer. The MMH and NTO are both very dense, stable and easily storable liquids. High density is considered desirable for liquid fuels, since it leads to lower mass through smaller tanks and less tank support structure for a given amount of potential thrust. MMH/NTO has an acceptable I_{sp} of 340 seconds[9], and is more stable than a pure hydrazine mono- or bi-propellant system.[15] One of the major advantages of an MMH/NTO system is its availability. Since this fuel combination is used extensively in the Space Shuttle, it is a logical assumption that the Space Station will have large MMH/NTO refueling tanks, which would eliminate having to waste Space Shuttle payload mass on fuel for the spacecraft. It may also be assumed that the pre-flight testing can be easily handled at the Space Station, since the Space Station crew will have experience in checking and testing the MMH/NTO Space Shuttle systems.

FUEL FEED SYSTEM

A pressure-fed fuel management system was chosen for the payload retro-rockets and the instrument bus. These two components use small systems that require minimal burn times. For these cases, the simplicity of pressure-feed is preferred. The Earth orbit escape and aerobraking fuel systems are considerably larger, so the thick tanks needed for pressure-feed would add considerable mass. A turbo-pump fuel feed system was chosen for its lighter mass and greater fuel flow capacity. The turbo-pump works by bleeding a small amount of gas exhaust and running it through a heat exchanger and turbine. The heat exchanger is wired to the fuel tanks so the required vapor pressure can be maintained as propellant is used. The turbine is used to power the actual fuel pump. The exhaust can then be sent back into the combustion chamber to boost performance with an afterburner effect, or can be expelled through a small gimballed nozzle to provide additional attitude control.[15]

PROPULSION SIZING

REQUIRED ΔV

The change in velocity needed to move a spacecraft to the desired location is called the ΔV. It is a direct function of initial and final masses of the spacecraft, and the I_{sp} of the specific propellant. It is given by Tsiolkofski's equation:

$$\Delta V = g_o I_{sp} \ln(M_i/M_f)$$ [7,15]

ΔV in m/s (ft/s)

g_o=9.8 m/s^2 (32.2 ft/s^2)

I_{sp}=340 s. for MMH/NTO

M_i=initial mass in kg (lb_m)

M_f=final mass (after burn) in kg (lb_m).

The propellant systems were broken down into four stages as listed:

--Stage 1 (Earth orbit escape): ΔV=3650 m/s from MP.
--Stage 2 (Aerobraking and vehicle AACS): ΔV=1470 m/s.
770 m/s for Aerobrake (AERO) [includes a very generous safety margin; 670 m/s would probably be sufficient.]
700 m/s for AACS [based on 20% of total propulsive ΔV;[12] considerably more than calculated in AACS section. Some extra fuel would be kept for contingency factor, however, the majority could be left out, which lessen overall spacecraft mass.]
--Satellite: fuel requirements calculated in AACS section.
--Retro-rockets (for payload): 75 m/s from AERO.

ΔV CALCULATIONS

In making these calculations, an average tankage factor of 15% of fuel mass was assumed. The tankage mass includes tanks, plumbing, pumps, etc. This figure may be high, considering recent advances in materials science. Structural masses are from Structures subsystem (STRU). Engine masses are approximations. The large engines are 6000 lb_f orbital maneuvering units from the Space Shuttle (off-the-shelf). The smaller engines are 900 lb_f thrusters, also from the Space Shuttle (off-the-shelf).[15]

Retro-rockets: 75 m/s=9.8 m/s^2 x 340 s x ln(M_i/M_f)

M_i=2300 kg payload re-entry vehicle	M_f=2617 kg initial mass
296 kg battery pack	60 kg propellant
12 kg retro engines (3 @ 4 kg)	2677 kg total
9 kg tankage	
2617 kg total	

Satellite: See AACS section for calculations. PPS was given a figure of 285 kg for fuel and tankage.

Stage 2: 1470 m/s=9.8 m/s^2 x 340 s x ln(M_i/M_f)

M_i=2677 kg total re-entry stage	M_f=4263 kg initial mass
500 kg aerobrake system	2378 kg propellant
285 kg satellite propulsion	6641 kg total
400 kg empty satellite	
4 kg satellite engine	
40 kg stage 2 engine	
357 kg stage 2 tankage	
4263 kg total	

Stage 1: 3650 m/s=9.8m/s^2 x 340 s x ln(M_i/M_f)

M_i=6641 kg mass into aerobraking	M_f= 9638 kg initial mass
120 kg engines (3 @ 40 kg)	19184 kg propellant
2877 kg stage 1 tankage	28822 kg total
9638 kg total	

TOTAL SPACECRAFT MASS = 28,822 KG.

TANK SIZING

The density of MMH=870.1 kg/m^3; NTO=1431 kg/m^3. For the pressure-feed systems, the mass ratio of MMH to NTO is 1:1.6. For the pump-feed stages, the mass ratio of MMH to NTO is 1:2.[9] Spherical tanks were chosen as being best for pressure control, as well as industry standard. A sample calculation and tank sizes follow:

Retro-rockets: 60 kg total propellant
60 kg / 2.6 parts = 23.1 kg/part
MMH = 23.1 kg (23.1 kg/part x 1.0 part)
NTO = 36.9 kg (23.1 kg/part x 1.6 parts)

23.1 kg MMH / 870.1 kg/m^3 = .0265 m^3
36.9 kg NTO / 1431 kg/m^3 = .0258 m^3

Volume of a sphere = $4\Pi r^3/3$ where r = radius.
MMH r = $(.0265 \times .75 / \Pi)^{1/3}$ = .185 m
NTO r = $(.0258 \times .75 / \Pi)^{1/3}$ = .183 m

	MMH	NTO
Retro tanks (Pressure-fed)	23.1 kg 1● r=.185 m	36.9 kg 1● r=.183 m
Satellite tanks (Pressure-fed)	99.6 kg 1● r=.30 m	159.4 kg 1● r=.30 m
Stage 2 tanks (Pump-fed)	792.7 kg 2● r=.48 m	1585.3 kg 2● r=.51 m
Stage 1 tanks (Pump-fed)	6394.7 kg 2● r=.96 m	12789.3 kg 2● r=1.02 m

THRUSTERS

Thruster sizing, selection and placement fell under the jurisdiction of AACS for the ACME vehicle. PPS and AACS determined that three-axis stabilization was best able to meet the needs of the various subsystems. PPS required three-axis stabilization since liquid fuel can cause quick stability deterioration in a spinning configuration due to fuel slosh.[12] Five and twenty-five Newton thrusters were chosen for their reliability and their off-the-shelf status. See AACS for more information.

POWER

SOLAR VS. NUCLEAR VS. CHEMICAL

Solar arrays combined with chemical batteries were determined to be the best option for spacecraft power for this mission. Radioisotope thermal generators (RTGs) were eliminated due to safety and political reasons, along with the problem of placement. RTGs would need to be mounted on a boom so they would not affect Science (SCI) and CCC instrumentation. However, this boom would need to be retracted during aerobraking, and the close proximity might interfere with computer commands needed during this stage. Solar dynamic collectors would be too large for so small a satellite, and would be of questionable maturity at the 1998 technology deadline. Regenerative fuel cells (RFCs) carry too large of a mass penalty, and again, would be of questionable maturity.

The chosen power system consists of 2 flexible blanket solar arrays,mounted on the instrument bus, and a battery of nickel-hydrogen (Ni-H) 100 Amp-hour (Ah) cells in the payload re-entry vehicle, with a few additional cells in the instrument bus. The solar arrays use state-of-the-art silicon cells with an efficiency of 14%.[11] The Ni-H battery cells are U.S Air Force advanced design, with a mass of 3.70 kg/cell and depth-of-discharge (DOD) of 80%. These Ni-H are cylindrical, with a length of 13.64" and a diameter of 4.5" (as opposed to the standard 3.5").[8] These cells are currently going through pre-certification testing, and are expected to become standard in flight by the early 1990's.

POWER SIZING

To size the solar arrays and Ni-H batteries, the mission was divided into the following three modes:

--Earth to Mars: Solar arrays will be deployed throughout this mode. Eclipse times are negligible.

--Aerobraking: Solar arrays will be retracted for duration of aerobraking stage. All power will be supplied by Ni-H battery located in payload re-entry vehicle.

--Mars orbits: Solar arrays will be deployed. Power needed during eclipse times will be stored in small Ni-H battery in instrument bus.

SOLAR ARRAY SIZING (Earth to Mars Mode)

Solar arrays can be sized using the following formula:

$$\underline{P_{BOL}=S \times Cr \times e \times A \times (1-\alpha(t-25))}$$ [7]

P_{BOL}=Power at beginning of life (W)

S=solar constant at a given distance from the Sun (W/m^2)

[S=1353 w/m^2 at Earth, S=582.8W/m^2][13,17]

Cr=concentration ratio of cells on array

[Cr =.85 to .92; used .88 as an average][13,17]

e=efficience of cells

[e=14% for cells used][11]

A= area of array (m^2)

α=temperature degradation factor

[average α =.001 ([^{o} C]$^{-1}$)][13,17]

t=operating temperature of solar cells (^{o}C)

[t=60^{o}C at Earth, t=-5^{o}C at Mars][13,17]

P_{BOL} is found using the equation:

$$P_{TOTAL}=P_{BOL} \times [1\text{-time degradation factor(TDF)}]$$ [7]

The TDF approximates performance drop over time, due primarily to radiation breakdown of solar cells. Assumed TDF of 6%/year.

Power loads for Earth to Mars mode:

250 W for CCC
145 W for AACS
<u>15 W for miscellaneous PPS</u>
410 W total

P_{BOL}=(410 W)/.70=586 W.

At Earth: P_{BOL}=1353 W/m^2 x .88 x .14 x A x (1-.001(60-25))=>A=3.64m^2

At Mars: P_{TOTAL}=586 W x (1-.06)=551 W

551 W=582.8 W/m^2 x .88 x .14 x A x (1-.001(-5-25))=>A=7.91m^2

BATTERY SIZING (Aerobraking Mode)

Batteries can be sized using the following equations:

$$\text{Stored Energy (SE)}=P_{LOAD} \times T_E / DOD$$ [7]

SE in Watt-hours (Wh)

T_E=time in eclipse (h)

Recall aerobrake battery cells are 3.7kg/cell x 55Wh/kg=>203.5 Wh/cell

Power loads for aerobrake mode:

125 W AACS (in atmosphere: see PFW-11)
30 W AACS (landing sensors-one day max.)
10 W CCC (for 15 minutes/cycle x 77 cycles)

SE=(125 W x 96.6 h + 10 W x 19.25 h + 30 W + 24 h)/.80 = 16235 Wh
Add 65 Wh as a safety margin for a total of 16300 Wh.
16300 Wh / 203.5 Wh/cell = 80 cells in the aerobrake battery.
80 cells x 3.7 kg/cell = 296 kg

COMBINED SIZING (Mars orbit mode; instrument bus only)

Solar arrays combined with battery storage can be sized using all equations above, and in addition:

$$P_{TOTAL}=P_{LOAD}+(C/N) \times V$$ [7]

C/N=charging current of batteries
C=battery capacity (Ah)
$N \leq T_s/DOD$ (h) [7]
T_s=time arrays are in sun/cycle (h)

Will be using two Mars orbits in this mission:

	Low Orbit	High Orbit
Altitude (mi)	248.5	10,607.5
T_e (h)	.691	1.306
T_s	1.270	23.354
Power load (W): AACS	250	115
CCC	100	100
SCI	95	35
N (h)	1.5875	29.1925
P_{BOL} (W) [P_T=P_T+10%]	3890	565
SE (Wh)//# cells	397.3//3	408//3

Final solar array size (total) = 52.6 m^2

(A=3890/(528.8 x .88 x .14 x [1-.001(-5-25)])

POWER DISTRIBUTION

Power will be distributed to the various load components through an unregulated power bus, since no component has a set voltage requirement. The unregulated bus has lower mass and is more reliable. All standard fault protection will be used (redundant fuses and wiring, etc.), since the added mass is relatively small. Some pre-flight testing will be done at the Space Station to assure that the distribution system and all safety measures are working properly.

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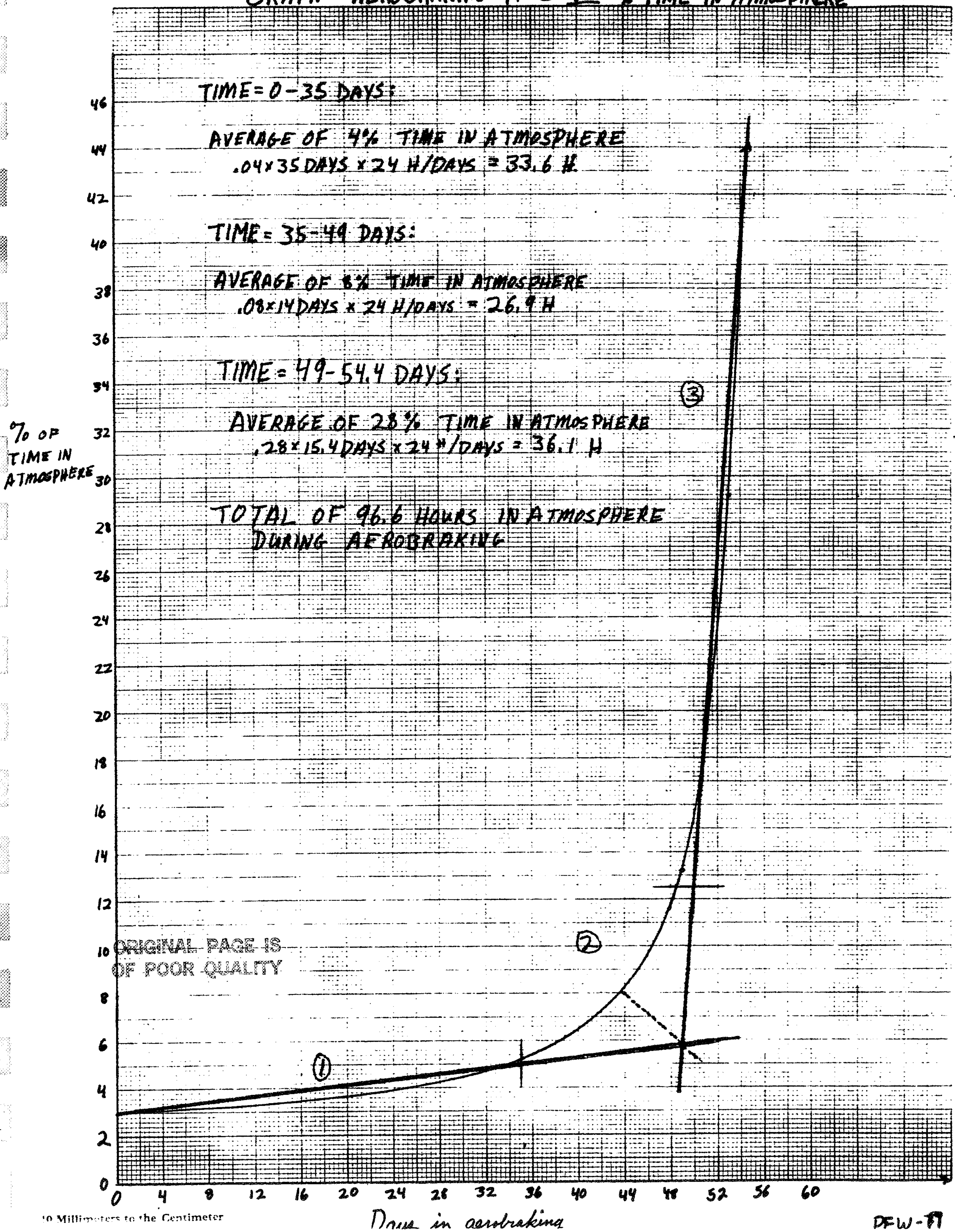
GRAPH: AEROBRAKING TIME VS % TIME IN ATMOSPHERE



ORIGINAL PAGE IS
OF POOR QUALITY

10 Millimeters to the Centimeter

SUBSYSTEM INTERACTION

MISSION PLANNING:

--ΔV NEEDED FROM M.P. TO DETERMINE AMOUNT OF FUEL NEEDED

COMMAND, CONTROL & COMMUNICATIONS:

--MUST BE ABLE TO SUPPLY ADEQUATE POWER TO COMPUTER AND COMMUNICATIONS EQUIPMENT AT ALL TIMES, INCLUDING AEROBRAKING PHASE

--MUST BE ABLE TO RECEIVE COMMANDS FROM COMPUTER ABOUT POWER DISTRIBUTION

ARTICULATION AND ATTITUDE CONTROL:

--MUST SUPPLY POWER TO ATTITUDE CONTROL DEVICES AND THRUSTER IGNITERS

--MUST USE AACS TO POINT SOLAR ARRAYS

--ATTITUDE CONTROL METHOD AND FUEL CHOICE HAVE STRONG LINK (LIQUID FUEL CAN QUICKLY DESTABILIZE SPIN-STABILIZED CRAFT)

STRUCTURES:

--MUST SUPPLY POWER FOR ANY ACTIVE THERMAL CONTROL SYSTEM

--THERMAL CONTROL NEEDED AROUND PROPULSION COMPONENTS

--STRUCTURE NEEDED TO SUPPORT PPS

--OVERALL SPACECRAFT STRUCTURE MUST BE ABLE TO SUPPORT FORCES CAUSED BY PROPULSION SYSTEM CHOSEN

--NUMBER OF PROPULSION STAGES AFFECTS OVERALL CONFIGURATION OF SPACECRAFT

AEROBRAKE:

--EFFICIENCY OF AEROBRAKE DIRECTLY AFFECTS AMOUNT OF FUEL NEEDED

--LENGTH OF TIME AEROBRAKING AFFECTS CHOICE OF POWER SOURCE FOR AEROBRAKE MODE

SCIENCE:

--MUST PROVIDE POWER NEEDED BY SCIENCE INSTRUMENTS AT ALL TIMES

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STRUCTURE AND THERMAL CONTROL SUBSYSTEM

SUBMITTED BY THOMAS M. STYBR [redacted]

The purpose of the structural and thermal control subsystem is to provide integrity and support for payloads and equipment, support loads induced by launch, aerobraking, and other manuevers, and provide a proper thermal envelope for the spacecraft to ensure proper operating conditions. To accomplish these tasks, a series of steps were followed which led to an educated selection of approaches and procedures to produce a viable design. This section of the Final Design Report will show the direction and decisions followed in producing this subsystem and the integration with the rest of the spacecraft.

The following will be discussed:

-Structural Configuration Design

-Materials Selection

-Structural Component Sizing

-Structural Components Summary

-Heat Management and Thermal Control

-Thermal Control Coatings

-Spacecraft Layout

-Data Bus Layout

-Space Shuttle Compatability.

STRUCTURAL CONFIGURATION-LOAD BEARING

TYPE	ADVANTAGES	DISADVANTAGES
-Monocouque plate structure	-inexpensive -easily fabricated -excellent interior protection	-heavy -must be bulky to support loads
-Beam supported plate	-fairly inexpensive -good interior protection	-very heavy -restricts interior utilization
-Tubular truss skeleton supporting various coverings	-very light -high stiffness and dimensional stability -tubing provides near optimum geometrical structural properties -cost expected to drop	-fairly complex -moderate to high expense

The tubular truss skeleton will be used because of its superior light weight and dropping expense. The truss will also be very manageable in low Earth orbit for assemblage.

STRUCTURAL TUBING MATERIALS

Tubing	Strength: Weight	CTE*	Cost	Technology	Advant.	Disadvant.
Al-Ti (alloys)	low	high	low	proven	easy fabrication	-heavy -poor damping
Metal-Matrix	mod to high	mod low	very high	as yet unaccepted	impact strength	-poor damping
Carbon Fiber Reinforced Plastic (CFRP)	high	low	mod high	becoming accepted	-vibration damping -high stiffness	-outgassing -environmentally unstable -poor impact strength
Al-clad CFRP	high	very low	mod to high	becoming widely accepted	-micro-crack resistant -exceptional damping -high stiffness -low outgassing	-average impact strength

*Coefficient of Thermal Expansion CHOSEN:

Tubes would be adhesively bonded into Ti-6Al-4V alloy fittings which are machined and etched for maximum adhesive contact. The fittings would then accept sheet or covering mounts, appendages, and hardpoints such as trunions or manipulator arm grip-points. The alloy fittings will transmit loads from tube to tube. They will also accept pyro technic devices for stage and module separation.

STRUCTURAL MEMBER SIZING

The load requirement for tubes under tension or compression is:

$$Pcr = EI\pi^2/L^2$$

Pcr is the buckling load

E is the tube longitudinal modulus

I is the minimum moment of inertia of cross-section

L is the length of the tube

The required EI of the considered tube would then be equal to:

$$EI = Pcr\ L^2/\pi^2$$

The longest load bearing member is 16.2 feet long. This member is entered into the equation along with a axial thrust load (from propulsion) of 18,000 pounds force. The equation yields a required EI of 76,000,000pound-inches . From Ref. 1, Fig. 3, a tube with an inner radius of one inch would require a wall thickness of 400 mils. Weight per length is aquired from Ref.1, Fig. 4 and yields 1.6 lb/ft. These tubes are primary load bearers and are few in number. The same procedure yields secondary load bearers with 80 mil wall thickness and 0.41 lb/ft weight per length. Every tube is clad inside and out with 5 mils of aluminum.

Tubing manufactured from the P75 Graphite/Epoxy unidirectional composite material and clad with 6061 aluminum is inspected by the manufacturer for strength and consistency before aquisition. The processing has been proven and accepted as a reliable means of composite fabrication and further use of the process will result in lower costs and even greater reliability.

STRUCTURAL COMPONENTS SUMMARY

- REENTRY/PAYLOAD VEHICLE

- total mass - 874.5 kg -- 1923.9 lbm

-TUBING- 80mil P75 Gr/Ep, 5 mil Al cladding, 0.60kg/m -- 0.41 lb/ft

NUMBER OF TUBES	LENGTH OF TUBES	TOTALMASS
-10 X 12 @	1.20m 3.94ft	85.9kg 188.9lbm
- 9 X 12 @	2.00m 6.56ft	129.6kg 285.1lbm
- 14 @	2.33m 7.65ft	19.6kg 43.1lbm
- 2 X 4 @	3.32m 10.90ft	15.84kg 34.85lbm
- 2 X 4 @	2.32m 7.60ft	11.07kg 24.36lbm

-FITTINGS- Ti-6Al-4V

- 122 @	1.0kg (estimate) 2.2lbm	122.00kg 268.40lbm

-INSULATION- Fibrous Silica Felt (Advanced Flexible Reusable Surface insulation)

-2,620,800 cm^3 @ 0.16 $^{g}/_{cm^3}$ 419.3kg

922.5lbm

-THERMAL BLANKET- Fused Silica Fabric - 0.3cm thick

-262.08 m^3 @ 0.2718$^{kg}/_{m^2}$ 71.2kg

156.7lbm

-DATA BUS

- total mass - 257.66kg -- 566.9lbm

-TUBING - 80mil P75 Gr/Ep, 5mil Al cladding, 0.60kg/m -- 0.41 lb/ft

NUMBER OF TUBES		LENGTH OF TUBES	TOTAL MASS
-2 X 12	@	0.95m	13.68kg
		3.12ft	30.10lbm
- 12	@	1.50m	10.80kg
		4.92ft	23.76lbm
-2 X 4	@	1.75m	8.40kg
		5.74ft	18.50lbm
-2 X 4	@	2.60m	12.48kg
		8.53ft	27.46lbm

-SHEET - 100mil Aluminum-Lithium Alloy

- 12	@	(0.95 X 1.50 X 0.00254)m	
		@2300 $^{kg}/_{m^3}$	99.90kg
		(3.12 X 4.92 X 0.00833)ft	
		219.8lbm	

-FITTINGS - Ti-6Al-4V

- 24	@	1.0kg (estimate)	24.0kg
			52.8lbm

MISCELLANEOUS

-STRUCTURAL SUPPORT OF AEROSHIELD, PROPULSION, AND LANDING PACKAGES

-Sum of tubing @ 400mil wall thickness-->150.0m 110.0kg

-estimated number of fittings -->100 150.0kg

HEAT MANAGEMENT AND THERMAL CONTROL

Heat management and thermal control is accomplished utilizing existing technology with few modifications. The system design stresses passive control but a small active system is incorporated. The components chosen to maintain an acceptable thermal envelope for the spacecraft instruments and other components are :

-Heat Reservoir

-Bubble Injected Variable Conductance Heat Pipes (VCHP's)

-Instrument Integrated Heat Sinks

-Surface Mounted Solar Heat Collector

-Termostatically Controlled Variable Emmittance Radiator

The temperature dependant instruments will be indirectly coupled thermally by the VCHP network. When an instrument is operating, the heat it produces is managed by the integration of its heat sink and the VCHP. The circulation through the VCHP is managed by a low power active system consisting of a temperature sensor and a bubble pump. Excess heat is transmitted from the heat sink to the reservoir or heat is transmitted from the reservoir to the instrument as needed to maintain the programed temperature envelope.

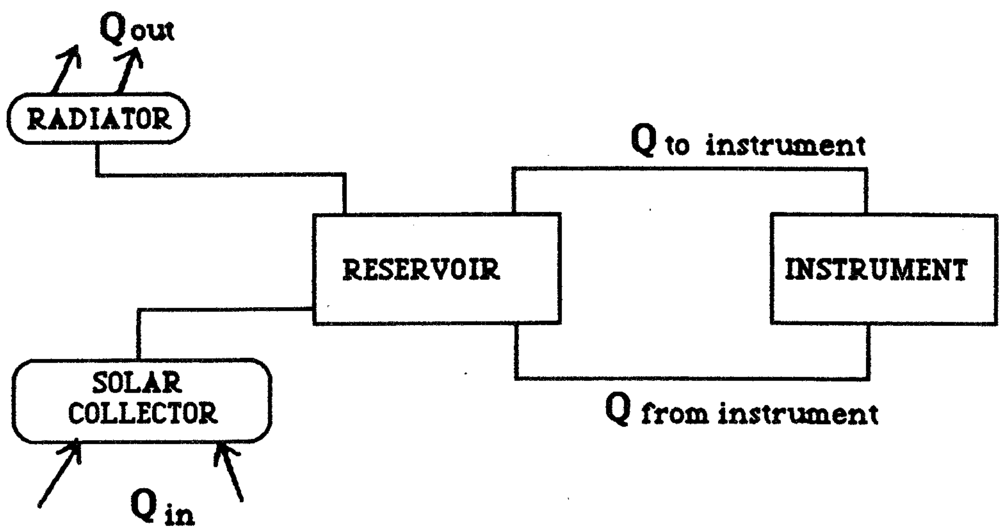
The reservoir acts as the system heat sink receiving heat from the instruments and solar collector and sending heat to the instruments and to the variable radiator. The VCHP's connecting the reservoir to the radiator and the solar collector are actively controlled to maintain the reservoir's heat amount. The reservoir's heat capacity is provided by a phase-change material (PCM) composed of inorganic salts which change phase from solid to liquid according to the heat content.

Heat is supplied to the reservoir by the solar collector. It is surface mounted on the data bus and uses its VCHP's working fluid as its working fluid. The fluid is then recycled back into the heat pipe. The collectors capacity is large enough to supply all the craft's components with enough heat to maintain temperature envelopes.

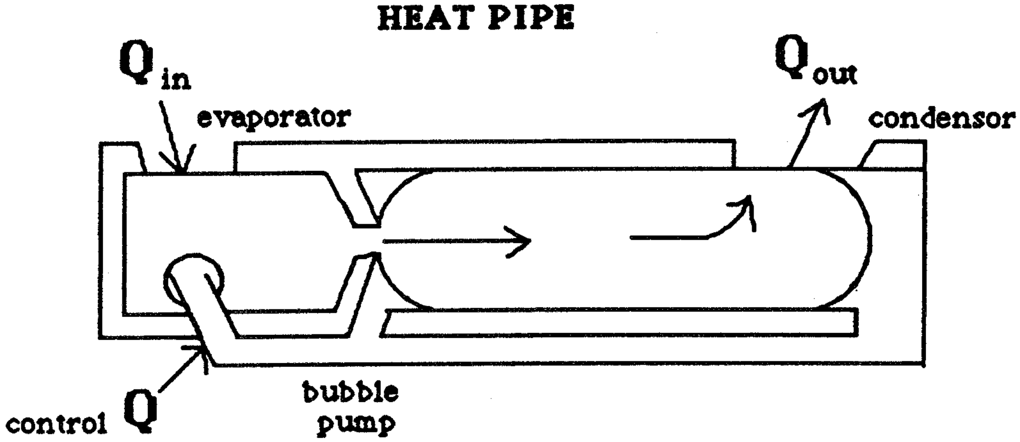
The variable radiator is controlled thermostatically with louvers. The reservoir's liquid portion of its PCM is circulated through the radiator at all times. When the heat becomes excessive, the thermostatic louvers open to allow heat radiation. At times of adequate heat content, the louvers remain closed.

The most complex portion of the system is the bubble injected VCHP. In zero gravity, the working fluid of the heat pipes need to be circulated by a means other than gravity. In the circulation of the fluid, normally ammonia, heat is transmitted by the vapor phase. Liquid ammonia is driven by a bubble pump into the evaporator where the liquid gains heat and vaporizes. The vapor has a very high vapor pressure as a result of the bubble pump and is injected into the pipe. It is this vapor pressure which drives the system.

THERMAL CONTROL SYSTEM SCHEMATIC



BUBBLE INJECTED VARIABLE CONDUCTANCE HEAT PIPE



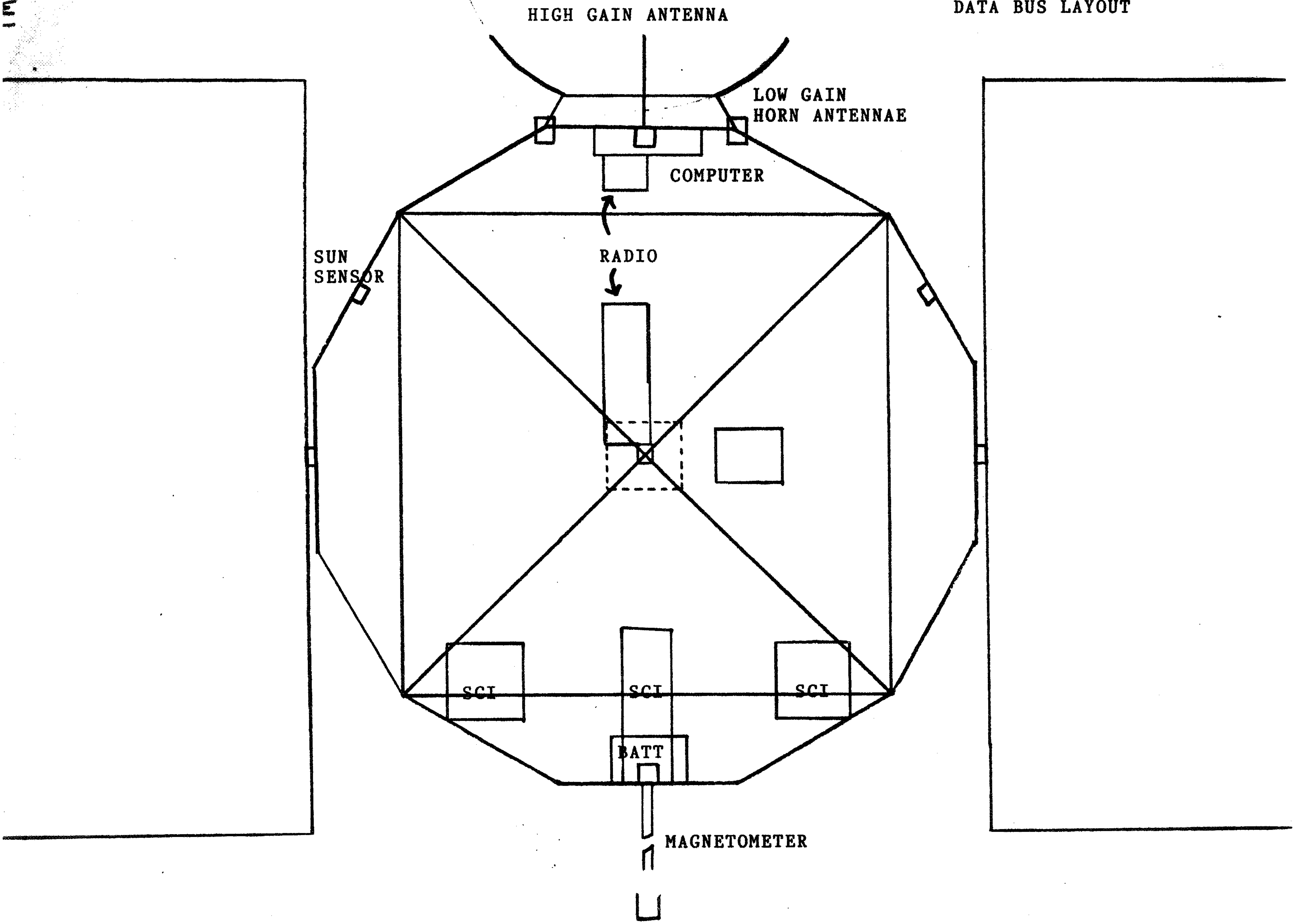
WORKING FLUID: AMMONIA

THERMAL CONTROL COATINGS

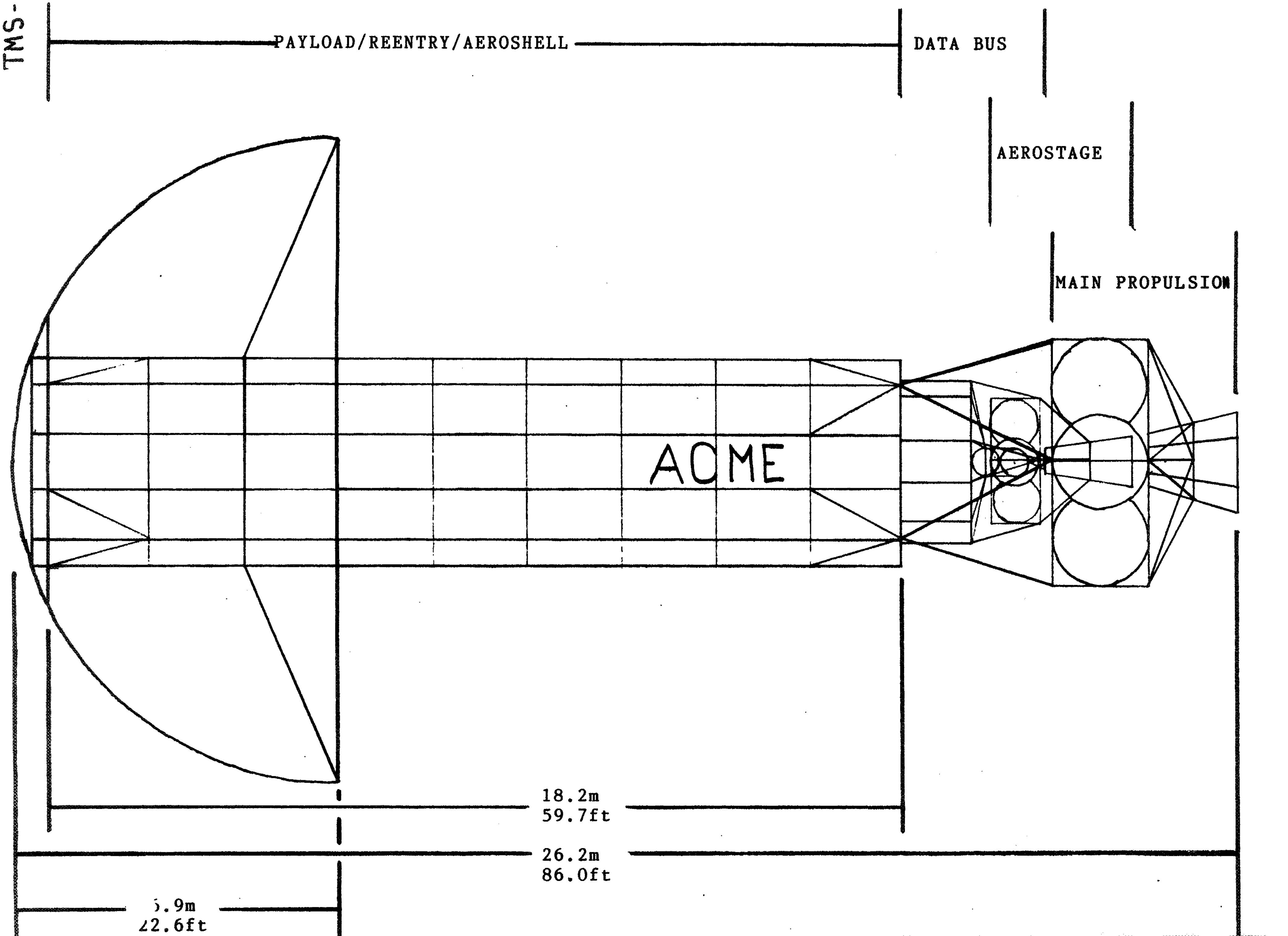
Zinc-Oxide Titanate on gray anodize	Silica Fabric with Carbon Yarn
-For use on propellant tanks -Features low absorbtance, low emittance -moderately easy application -Not for structural coatings	-For use as thermal control coating over large areas -Features tailorable properties with carbon yarn content -Easy application -Very durable, little degradation

These coatings have been chosen because of their acceptance and versatility although the silica fabric is just now being used widely. The Zinc/Titanium oxide coatings will be used to maintain a consistent temperature in the propellant tanks. The silica blanket will shield the structure from solar incidence and provide particulate protection.

DATA BUS LAYOUT



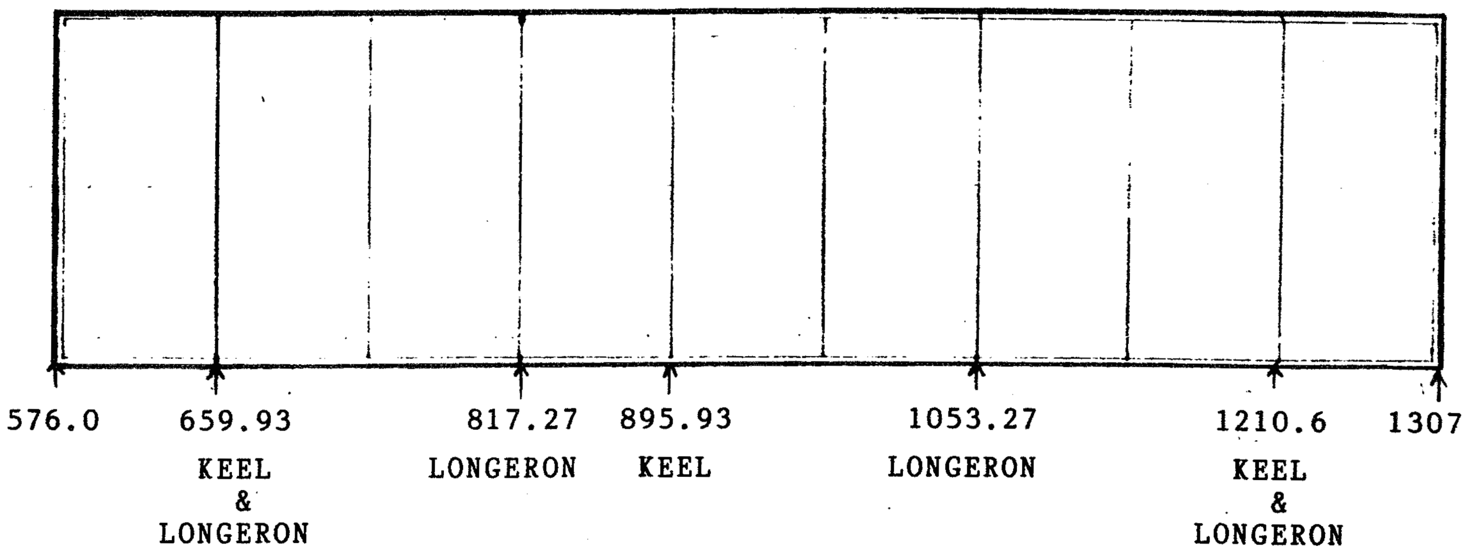
TMS-11


PAYLOAD/REENTRY/AEROSHELL
DATA BUS
AEROSTAGE
MAIN PROPULSION
ACME
18.2m
59.7ft
26.2m
86.0ft
.9m
2.6ft

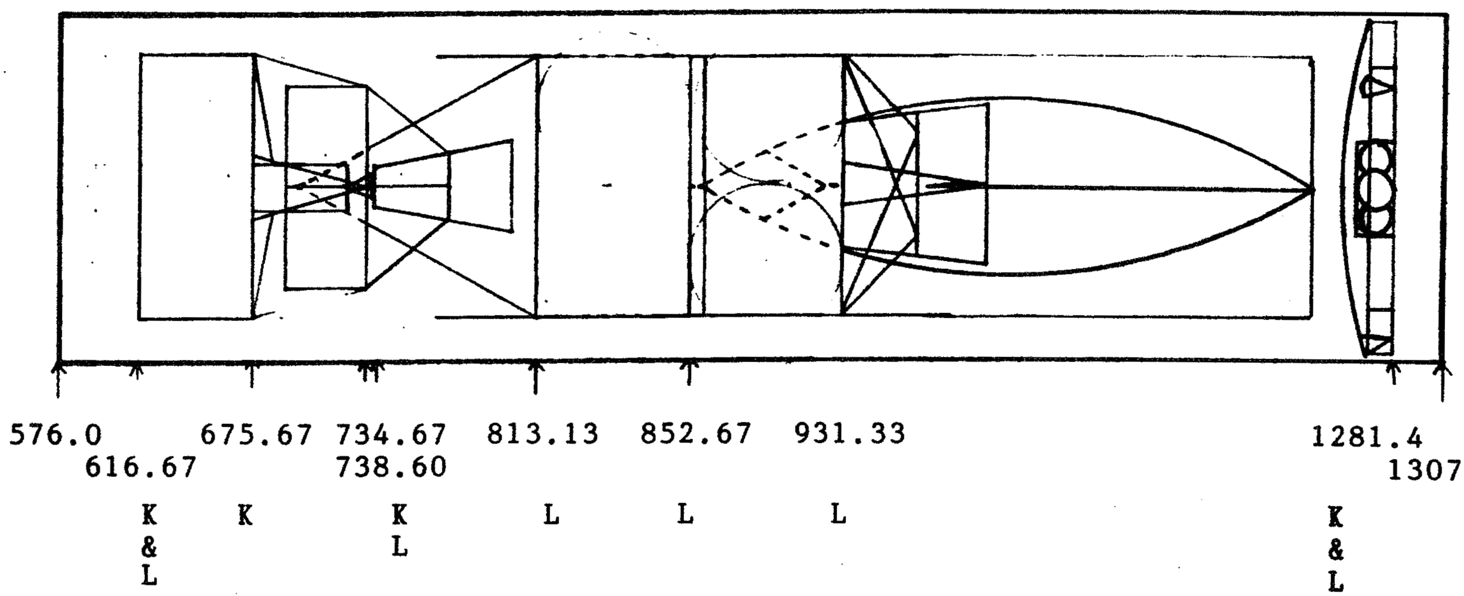
SPACE SHUTTLE

PAYLOAD BAY COMPATIBILITY

REENTRY MODULE



BUS/PROPULSION STAGES



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Aerobraking

by

Jay Onken

ss. [redacted]

Abstract: Method-of-attack, final parameters, and some problem areas are presented for the final design of the aerobrake subsystem for A.C.M.E.

REQUIREMENTS

In addition to the global requirements of the mission, the requirements for the aerobraking sub-system are:

1. Protect the payload (aircraft)
2. Dissipate orbit energy
3. Re-enter payload to Mars atmosphere
4. Safely land payload on Martian surface

Since aerobraking was a requirement for this mission, no data are given here to justify its use. It can be shown, however, that the use of aerobraking significantly reduces the amount of propellant mass needed to circularize an elliptic orbit.

SHAPE

It was decided early in the design process to use a hemispherical frontal shield for the aerobraking device. The other possibility was the use of a small frontal flat plate shield and an aft-deployable brake. Rationale for choosing the hemispherical frontal shield are outlined in Fig. AERO-1. It should be noted that simplicity and reliability, both being requirements in the RFP, are both "excellent" for the hemispherical shield.

SIZE

The size of the shield was determined mainly by the spacecraft length during aerobraking. The shield size also had to be optimized for an allocated aerobraking time of less than 2 months. A computer program called "Aerob" was utilized to optimize the aerobraking process. This program, developed by Stephen J. Hoffman, simplifies the aerobraking maneuver to optimize shield area, initial semi-major axis, and final mass in orbit for a given time limit. The inputs to the program include data for the Mars atmosphere plus some orbit and spacecraft data (see List AERO-1).

The smallest shield determined from the D-2D safety standard (see Fig AERO-2) had a radius of 6.9 - 7.0 m; therefore the flat plate area used in AEROB was approximately 150.0 m^2. Since the shield is hemispherical, however, the actual surface area of the shield will be 300.0 m^2. From this shield size, a 54 day time limit, an injected mass of 5000 kg, along with the other input parameters, the optimum initial semi-major axis was determined to be 95,000 km. (see Fig. AERO-3). With these parameters set, the final spacecraft mass in low-Mars-orbit is maximized at 4000 kg (see Fig. AERO-4); therefore, the propellant mass was minimized to circularize the orbit for these parameters.

COMPONENTS

The aerobrake components consist of a hemispherical frontal

Fig. AERO-1

SHIELD CRITERIA AND RATING CHART *

CRITERIA	RATING: HEMISPHERICAL FRONTAL SHIELD	RATING: SMALL FRONT SHIELD AND AFT-DEPLOYABLE AEROBRAKE
SUFFICIENT PAYLOAD PROTECTION	4	4
SIMPLICITY OF INTEGRATION	4	1
WEIGHT FACTOR	2	2
STABILITY	3	4
FEASIBILITY	4	3
RELIABILITY	4	2
OVER-ALL EFFICIENCY	4	2
TOTAL	25	18

RESULT:

HEMISPHERICAL FRONTAL SHIELD CHOSEN OVER AFT-DEPLOYABLE SHIELD

* RATING SYSTEM: 1-BAD
2-ACCEPTABLE
3-GOOD
4-EXCELLENT

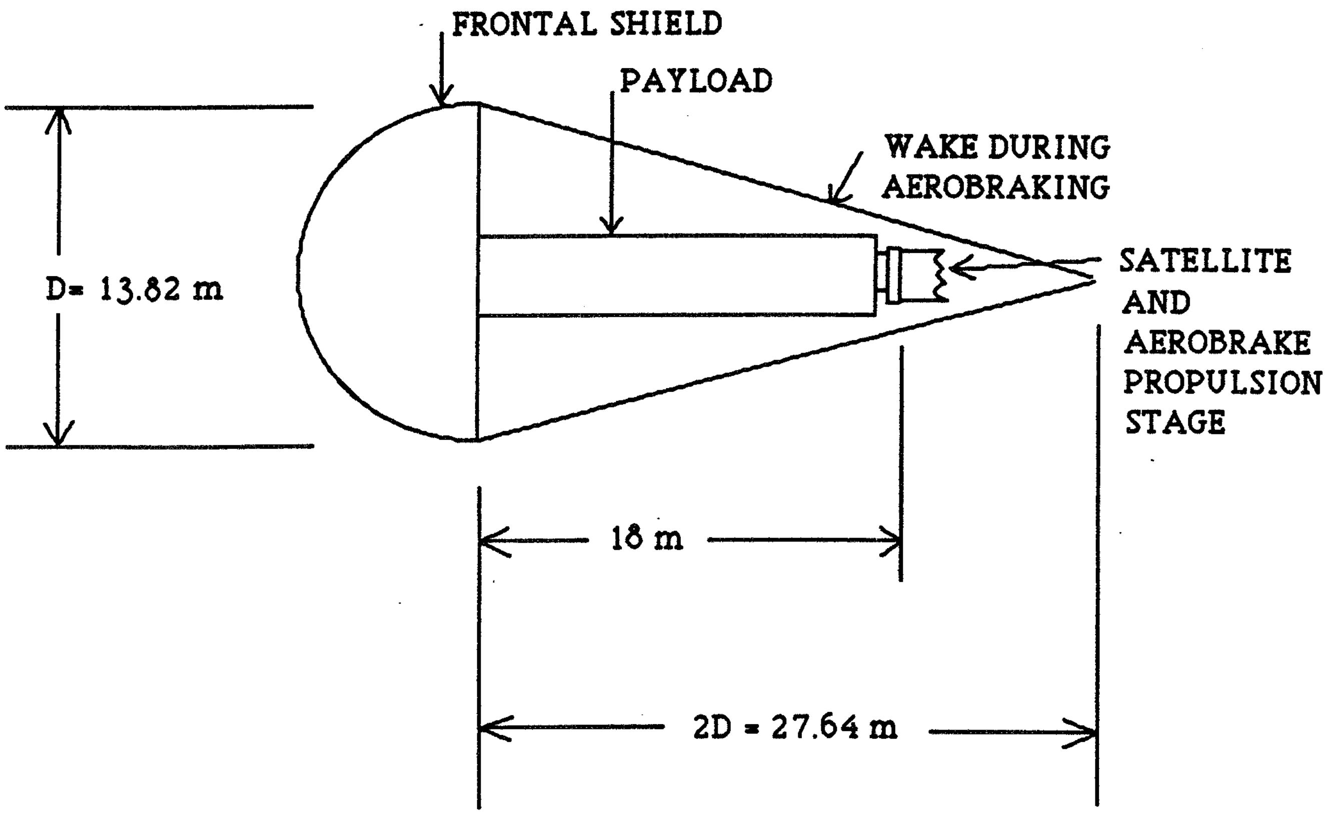
LIST AERO-1

AEROB INPUTS

SCMASS= 5000.0	= SPACECRAFT MASS (KG)
SCAREA=150.0	= SHIELD FRONTAL AREA (m2) (FLAT PLATE)
CD=2.0	=DRAG COEFFICIENT
ABWF=3.33	=AEROBRAKE WEIGHT FACTOR (kg/m2)
ISP1=380.0	=STAGE 1 Isp (sec)
TANKF1=0.15	=STAGE 1 TANKAGE FACTOR (tank mass/unit mass propellant)
ENGM1=0.0	=OTHER STAGE 1 INERT MASS (kg)
SJET1=0.0	=NO JETTISON OF STAGE 1
ISP2=340.0	=STAGE 2 Isp (sec)
TANKF2=0.15	=STAGE 2 TANKAGE FACTOR (tank mass/unit mass propellant)
ENGM2=0.0	=OTHER STAGE 2 INERT MASS (kg)
SJET2=0.0	=NO JETTISON OF STAGE 2
ISP3=340.0	=STAGE 3 Isp (sec)
TANKF3=0.15	=STAGE 3 TANKAGE FACTOR (tank mass/unit mass propellant)
ENGM3=0.0	=OTHER STAGE 3 INERT MASS (kg)
SJET3=0.0	=NO JETTISON OF STAGE 3
TMAX=400.0	=MAX. SHIELD TEMPERATURE (°K)
C = 450.0	=SPECIFIC HEAT OF SHIELD MATERIAL (J/kg/°K)
ALP= 0.7	=SHIELD ABSORPTIVITY
EPS= 0.7	=SHIELD EMMISIVITY
A= 95,000	=SEMI-MAJOR AXIS (km)
TIMLIM = 56.0	=MAXIMUM AEROBRAKING TIME (days)
ORBWAT = 0.0	=WAIT TIME IN INITIAL ORBIT BEFORE AEROBRAKING (days)
RPINT =3500.0	=INITIAL PERIAPSE RADIUS (km)
RFINAL = 3743.0	= FINAL ORBIT RADIUS (km)
PMIN = 0.08	=MINIMUM ORBIT PERIOD (days)
VINF = 2.5	= V_∞ ON HYPERBOLIC APPROACH (km/sec)

FIG. AERO-2

D-2D PAYLOAD PROTECTION REQUIREMENT



PAYLOAD PROTECTED FROM WAKE

PROTECTION REQUIREMENT MET

FIG. AERO-3

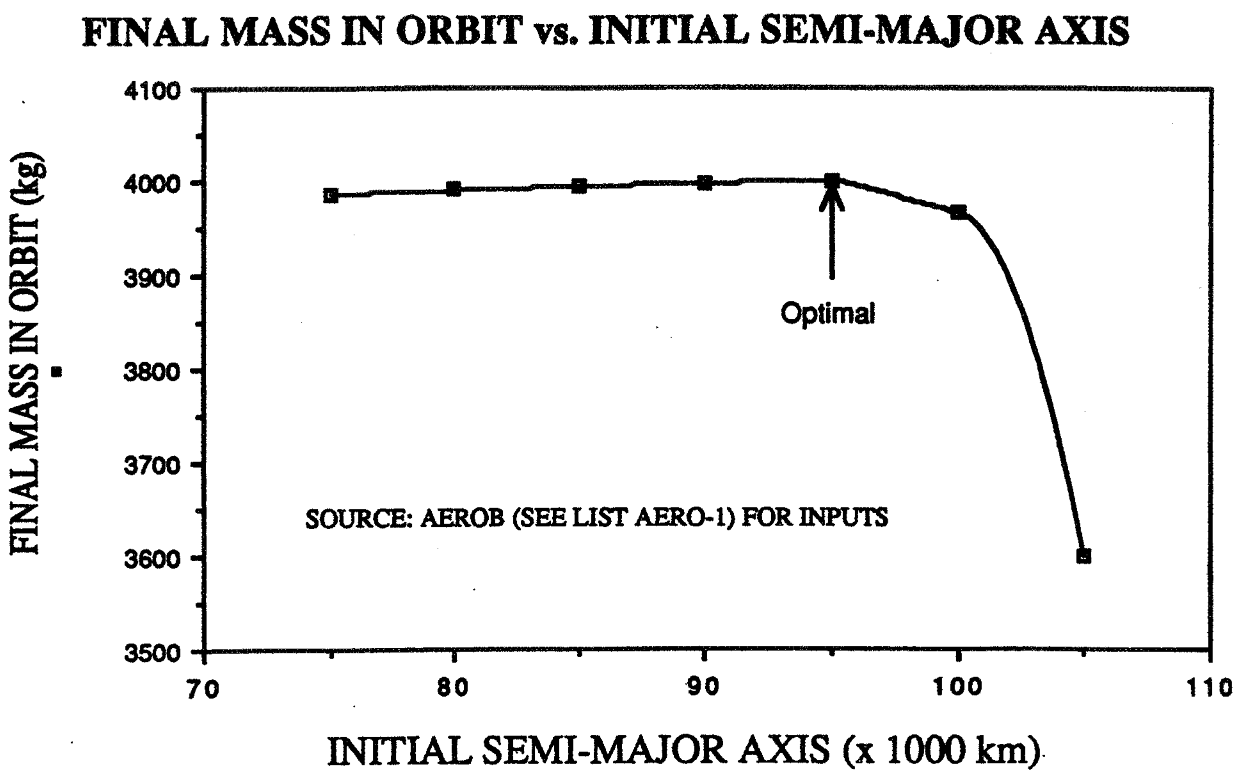
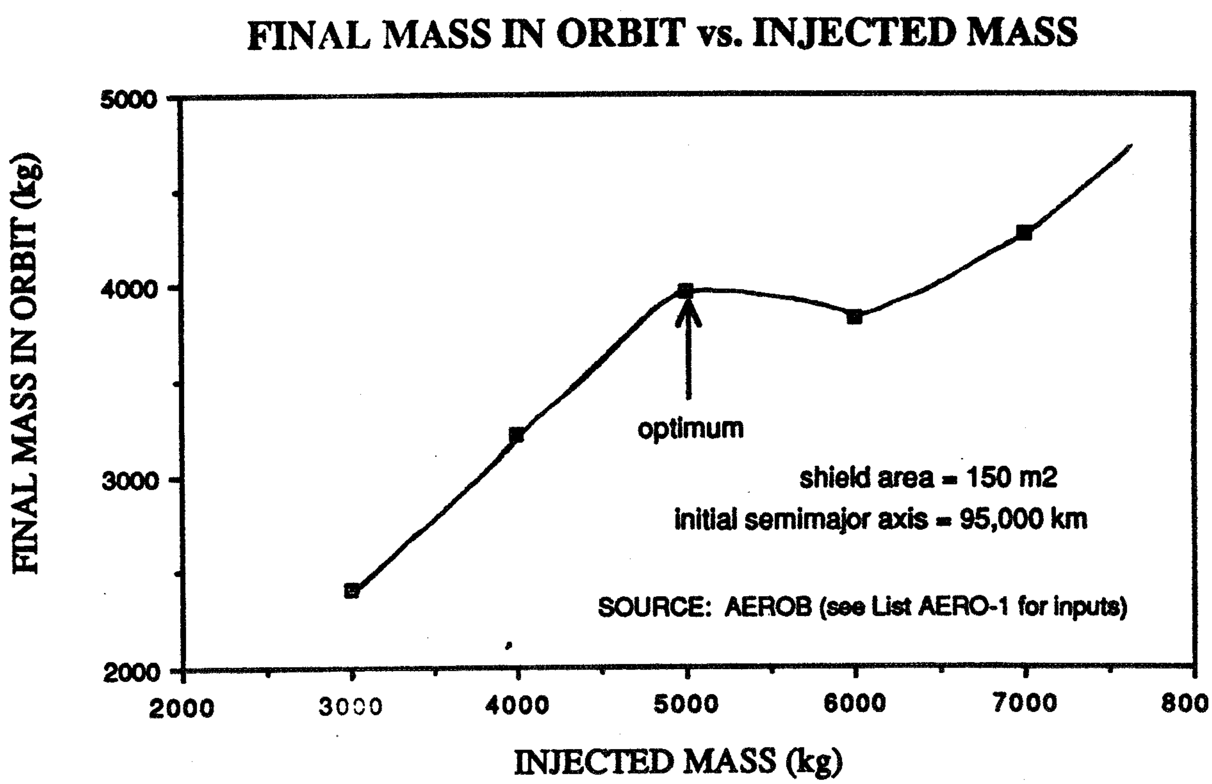


FIG. AERO-4



shield, landing gear and pads, an on-board computer, mortar assembly, parachute, retro-rockets, and control thrusters. The aerobrake shield is connected to the payload through a circular truss structure. The landing-gear is folded up against the payload until its deployment during re-entry. The on-board computer equipped with sensors is included to monitor the shield temperature and control the re-entry/landing sequence. The mortar assembly is located in the aft portion of the payload to be used for parachute deployment. Retro-rockets are located in the truss structure to slow the payload to its soft landing speed. Control thrusters are placed on the payload for control during aerobraking. (See spacecraft component layout -Structures.)

DYNAMICS AND CONTROL

For stability during aerobraking, the center of mass of the vehicle must be ahead of the meta-center.[1] Finding the meta-center is a technical problem that requires extensive knowledge of the properties of hypersonic flow on Mars; therefore, further study must be done to determine the extent of stability control needed. For this design, a worst-case scenario was assumed and control thrusters sized accordingly. See Attitude and Articulation Control Subsystem for sizing of the thrusters.

POWER REQUIREMENTS/SUPPLY

Since the solar panels will be retracted during aerobraking, power requirements to power the entire spacecraft during the allotted aerobraking time.These batteries will then power the on-board computer that will control the re-entry/landing sequence. See the Power and Propulsion Subsystem for sizing information.

COMMUNICATIONS

An umbilical cord connects the aerobrake to the satellite for transmission of communications and power before the separation sequence is begun. This cord is severed at separation and the re-entry vehicle acts independently until landing.

SEPARATION/RE-ENTRY/LANDING

The re-entry/landing sequence takes about 10 minutes and utilizes parachute and retro-rocket deceleration (See Fig. AERO-5). First, at a time to be determined by landing site conditions and orbit parameters, a pyro-technic device is fired to separate the satellite from the re-entry vehicle (REV). The REV is then lowered in orbit by 1km with the control thrusters. Then, the retro-rockets are fired through holes in the shield initially plugged with beryllium plugs to de-orbit the REV. Control thrusters will then orient the ship vetrically for re-entry. Free-fall will occur until the computer senses a height of around 7 km from the surface. At this point, the mortar assembly is fired to deploy the parachute, while the shield is released from the payload allowing it to fall away. This parachute, with an area of 1000 m^2, will slow the payload to a terminal velocity of 60 m/s (See Fig. AERO-6). Then, when the computer senses a height of around 1.5 km above the local terrain the retro-rockets are fired to slow the REV to a velocity of 2 m/s before landing. These retros are sized to slow the payload from 60 m/s to 2 m/s within 500 m at full thrust (See PPS and Fig. AERO-7). This leaves about a 1 km safety margin in case problems are encountered. At touchdown the retros are automatically shut off by switches in the landing pads. This sequence is similar to that used for the Viking lander[2]; therefore, it has been proven to work and utilizes

Fig. AERO-5

RE-ENTRY /LANDING SEQUENCE

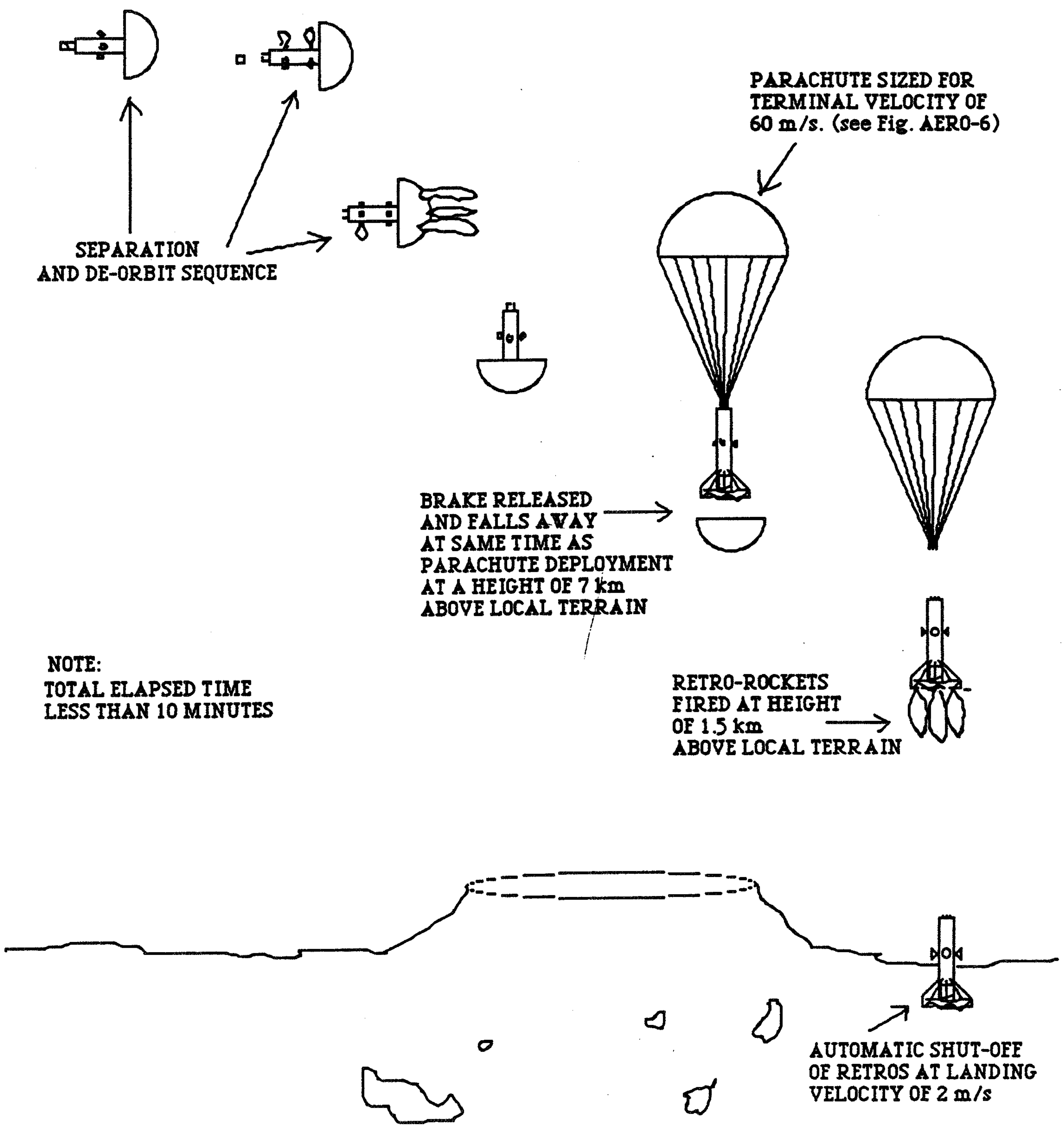
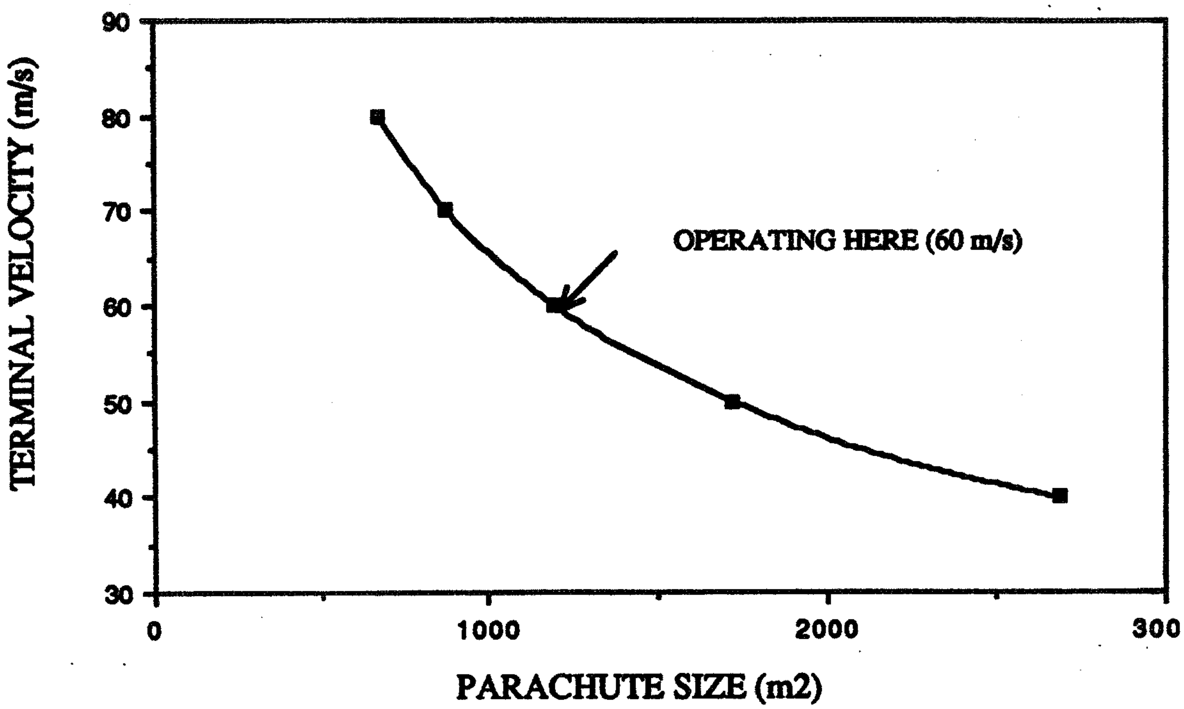


FIG. AERO-6

TERMINAL VELOCITY vs. PARACHUTE SIZE



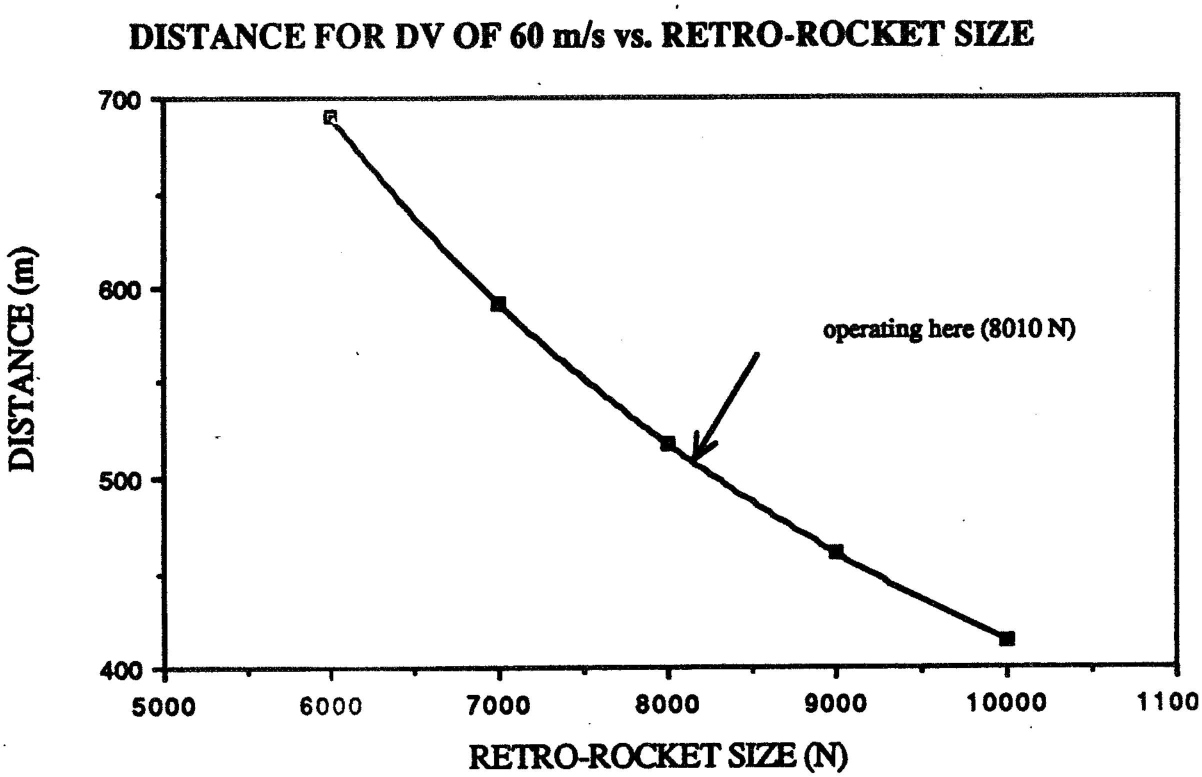
* Parachute equation

$$V_{TERMINAL} = \sqrt{\frac{2W}{P\, C_D AREA}}$$

W=WEIGHT (N) = 8339.8 N
P=DENSITY OF THE ATMOSPEHRE (kg/m^3)
(TAKEN HERE TO BE AT ALTITUDE OF 10 km)
C_D= CO-EFFICIENT OF DRAG = .6
(MEAN VALUE FOR VIKING LANDER PARACHUTE)
AREA= CROSS-SECTIONAL AREA OF THE PARACHUTE (m^2)

* source: Brown, W.D. , Parachutes, Pittman and Sons, New York, 1957. p.7

FIG. AERO-7



EQUATION: $\Delta X = \frac{M\,DV^2}{2F}$

WHERE: ΔX= DISTANCE TO ATTAIN DV OF 60 m/s
M= MASS OF THE REV = 2300 kg
DV= CHANGE IN VELOCITY = 60 m/s
F = FORCE EXERTED BY RETRO-THRUSTERS

"off-the-shelf technology." The parachute could be a problem area, though, and needs to be tested in the Earth's upper atmosphere before it is implemented on Mars.

SUBSYSTEM INTERACTIONS

The aerobraking subsystem interacts directly with Mission Planning, Structures, AACS, and PPS subsystems. (See Fig. AERO-8.) It should also be noted that there is indirect interaction with the Communications subsystem, because a low-gain antenna had to be used for communications during the aerobraking maneuver. (See Communications, Command, and Control subsystem.)

References

1 S.M. Yen, personal communication.

2 Viking Lander "As Built" Performance Capabilities. Martin Marietta, pp. I-21-26.

FIG. AERO-8

SUBSYSTEM INTERACTIONS

SUBSYSTEM	INPUTS TO AEROBRAKING	AEROBRAKING INPUTS	RESULTING DESIGN OUTPUT
MISSION PLANNING	TIME ALLOTMENT APPROACH SPEED FINAL ORBIT RADIUS	AEROB INPUTS	OPTIMUM INITIAL SEMI-MAJOR AXIS
STRUCTURES	SPACECRAFT LENGTH ATTACHMENT METHOD	D-2D REQUIREMENT —	SHIELD DIAMETER DETACHMENT METHOD
AACS	—	DYNAMICS PROBLEM	CONTROL THRUSTER SIZING AND PLACEMENT
PPS	MASSES	FINAL TIME FOR AEROBRAKING ΔV FOR LANDING RETROS	BATTERY SIZING RETRO SIZING

1.0 THE ACME ATTITUDE AND ARTICULATION CONTROL SYSTEM

By Eric Olsen

[redacted]

1.1 Introduction

The Acme attitude and articulation control system(aacs) will provide control to both the delivery vehicle and instrument bus, both of which are three-axis stabilized. The delivery vehicle uses only reaction control jets during the orbit to Mars and during aerobraking, and uses the retro rockets for attitude control during descent. The satellite will use three orthogonal reaction wheels aligned parallel to the principle axes.

2.0 Methodology

The general methodology utilized in the design of the aacs, for both the delivery vehicle and satellite, consisted of the use of trade studies to determine the mode of control. A matrix of the possible sensors was established, and based on the accuracy requirements, sensor weight and power, and mode of control, the equipment used on the vehicles was determined. The placement was chosen so that the sensors were able to operate. The sizing of the wheels, and thruster size/location was based on established performance criteria, which is discussed in the appropriate subsection.

3.0 Summary of Requirements

The general requirements of the aacs are listed under the global requirements section.

The specific requirements of the aacs are:

- to send telemetry to c^3
- to accept commands from c^3
- to receive power from pps
- to maintain satellite attitude to an accuracy of 0.1 deg in roll and pitch, and 0.3 deg in yaw
- to maintain the delivery vehicle accuracy to allow for communication to Earth(maximum total attitude error of 0.4 deg)
- to control the s/c and satellite attitude, guidance and navigation
- delta-v trims(station keeping)
- solar array pointing
- valve actuation
- reception of sensor inputs
- to allow for proper operation of the scientific instruments.

4.0 Delivery Vehicle Control System

Three-axis stabilization was chosen for the attitude control mode for the delivery vehicle. An initial trade study was conducted to determine the optimum method between CMG's and pure jet. CMG's were discarded because of the limited duration of the delivery vehicle mission(374 days), and their large mass.

4.1 Fuel Requirement for the Delivery Vehicle Control System.

The fuel requirement for the delivery vehicle derived here includes only the fuel needed for the control thrusters. The fuel needed by the retro-rocket package is calculated in the propulsion section. The estimate is derived by assuming the fuel needed is approximately equal to the fuel needed to rotate the s/c 360 deg,once a day.
$2(5N)(2.25m)=(24855.68kgm^2)\alpha$, $\alpha=9.052x10^{-4}rad/s^2$. It is accelerated to 2 deg/s, so $2deg/s(\pi/180)/\alpha=38.56sec$. The rotation is assumed to be about the longitudinal axis.

The Impulse needed from Earth to Mars(excluding aerobraking, flight time of 320 days) is, $=38.65sec(2)(10N)320=2.46x10^5Ns$

The impulse needed by the delivery vehicle aacs during the aerobraking phase is calculated similiarly, except $I_{xx}=18180kgm^2$

The flight time is about 54 days, so $2(5N)(2.25m)=I_{xx}\alpha$,

$\alpha=1.24x10^{-3}r/s$, $Impulse=28.21sec/day(10N)2(54days)=3.047x10^4NS$

Total aacs fuel requirement for the delivery vehicle is
$3.047X10^4+2.46x10^5Ns/((340s)(9.81m/s^2)=82.9kg$

4.2 Delivery Vehicle AACS Equipment Location and Characterstics

A trade study was conducted to determine the sensors used on both the satellite and delivery vehicle. The following matrix was established for use during the trade study.

sensor	mode		accuracy		
	spinning s/c	3-axis stab	low	med	high
IR static		*		*	*
IR scanning		*		*	*
High acc sun		*			*
IR pencil type	*		*	*	
star mapper	*			*	*
star tracker		*			*
fan beam sun	*			*	

Based on the mission requirements, sensor weight and power, mode of operation ,and accuracy, the following instruments are used on delivery vehicle.

It is assumed that the control system will be linked to the computer on the aircraft.

no per s/c	component	mass	power
1	star tracker triad	18.18kg	20w
1	hexad strapdown unit	20.72kg	125w
1	radar/altimeter	15kg	50w

The location of the equipment is outlined in the structures section.

Star Tracker Triad
This unit provides inertial attitude reference during the orbit to Mars and during part of the aerobraking procedure(the unit is off when the vehicle is in the atmosphere). Redundancy provided.
mass=18.18kg power consumption=20W accuracy=.005 deg

Hexad Strapdown Unit
This unit consists of accelerometers and gyroscopes with appropriate redundancy. This provides attitude and rate information about all three axes. The unit operates when the vehicle is in the atmosphere (aerobraking) and during the landing phase.
mass=20.72kg power consumption=125W

Radar
Used to provide altitude reference during the landing phase.
mass= 15kg power consumption=50w

4.3Operational Modes

Cruise Mode
This consists of the orbit from Earth to Mars. Attitude reference is provided solely by the star tracker triad. The Hexad unit is off. The delivery vehicle is three-axis active stabilized.

Aerobraking Mode
When the delivery vehicle enters the atmosphere, attitude reference is provided by the hexad unit, which is in operation whenever the vehicle is in atmosphere. Attitude control is provided by the control thrusters.

Seperation Mode
The initial orientation of the instrument bus is provided by the delivery vehicle. After seperation, the delivery vehicle will use the aacs jets to translate about 1 km. away from the satellite so that the retro rockets may be fired to initiate the landing phase.

Landing Phase
see aerobraking section.

4.4Delivery Vehicle Thruster Location and Characteristics

The thrusters are located to provide control about all three axes. Additionally, it will be required to translate(1 km) the delivery vehicle using the aacs thrusters. A feasibility study was done to see if the aacs control thrusters could provide a sizeable degree of control during landing. Due to the large mass of the delivery vehicle, attitude control will be maintained solely by the retro rockets during terminal descent. The sizing of the control thrusters was determined based on the desired response and an estimate of the degree of control the aacs will have to provide while the vehicle is aerobraking.

The thrusters are designed to accelerate the vehicle to 2 deg/sec in approximately one minute.

$2\text{deg/sec}(\pi/180)(1/60\text{sec})=5.817\times10^{-4}\text{rad/s}^2=\alpha$

$2(F)(2.25m)=I_{xx}\alpha,\ I_{xx}=24855.67\text{kgm}^2,\ F=3.21N$

The NASA standard 5N thruster will meet this requirement, and will be able to provide a fine degree of control due to its minimum achievable impulse time. In addition to the 12 5N thrusters for attitude control there will be an additional 8 thrusters (see Fig 1.) which will be used for the translation maneuver, and which can provide additional control authority which may be needed during aerobraking. These thrusters are designed to accelerate the vehicle to approximately 1m/s in 1 min. $a=1\text{m/s}/60\text{ sec}=.0167\text{m/s}^2$

$2(F)=M_{sc}a$, where $M_{sc}=3000.16$kg, $F=25.0N$

A 25 N thruster was chosen. This will enable the vehicle to perform the maneuver in a short perion of time. Thruster location is shown on fig 1. It is assumed that the thrusters will provide stability during the aerobraking phase. An exact analysis of expected disturbance due to the Marsian atmosphere may result in a change of thruster strength.

4.5Interaction of Delivery Vehicle AACS With Other Subsystems

The delivery vehicle aacs interacts with:

i)**Aerobraking Subsystem**- primary interaction is concerned with the delivery vehicle stability during aerobraking and descent. Further analysis should look at the feasibility of using flaps for control. An accurate estimation of the disturbing torques due to the atmosphere needs to be determined. Thrusters descibed above provide control during aerobraking.

ii)**C^3** -primary interaction concerned with the control of the antenna systems. The Acme antenna has the ability to point independent of vehicle attitude.(see c^3 subsection for description) The delivery vehicle aacs is linked to aircraft computer.

iii)**Science**-see satellite aacs/science interaction.

iv)**Structure Subsystem**-interaction concerned with the placement of aacs components to meet requirements. Additionally, s/c configuration was made so that inertia characteristics were optimized for control. See structures subsection for component layout.

v)**PPS**-all power requirements and fuel requirements described are provided by pps(see pps section for further details)

4.6Scanning and Pointing Requirements Implementation

see communications section.

4.7Pre-launch Launch Tests

Pre-launch tests will include test firing all control thrusters. An operations test will also be conducted on all Sensors.The satellite attitude electronics, and thrusters also tested.

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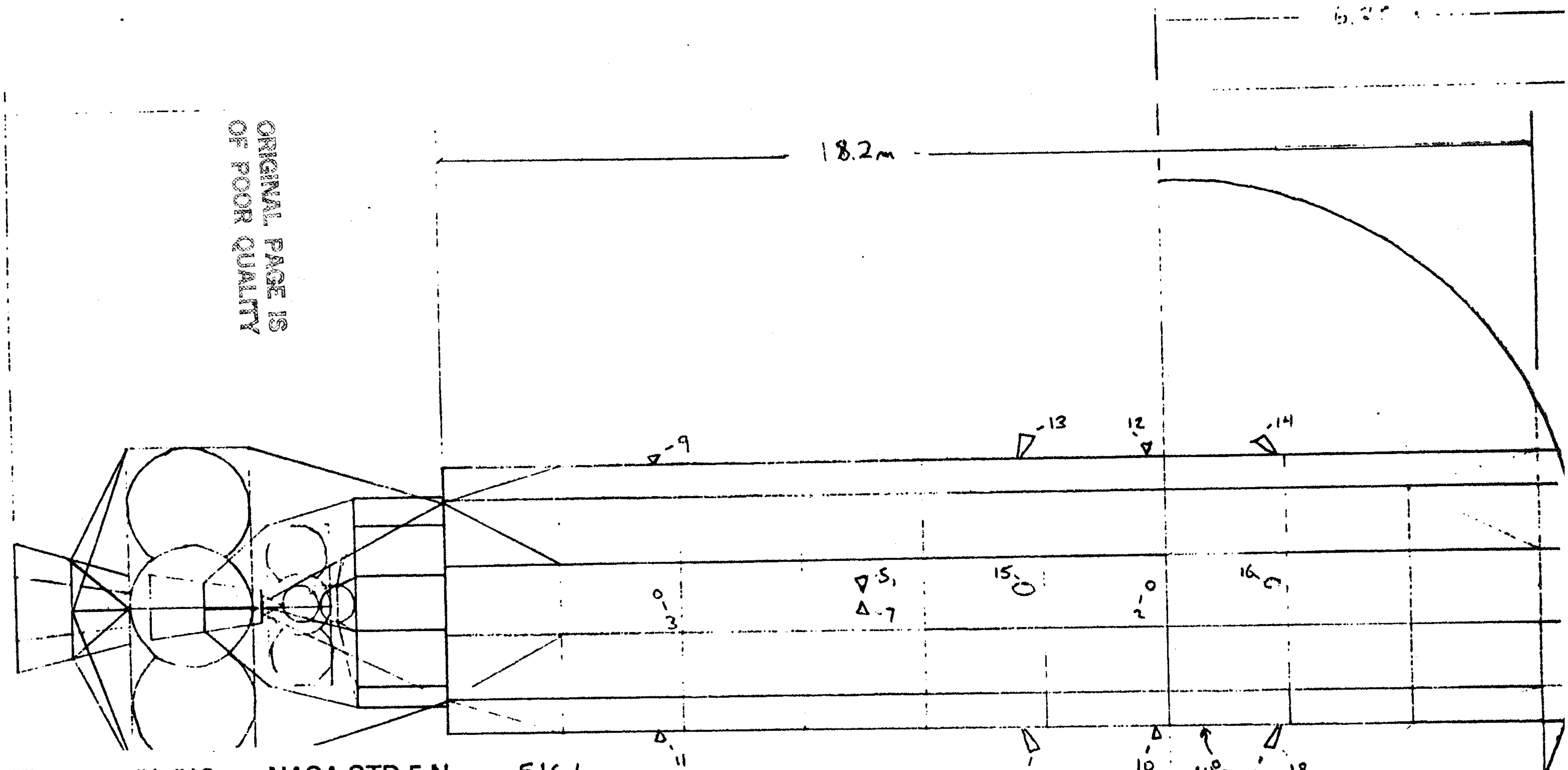

FIG 1.

Thrusters #1-#12 are NASA STD 5 N
Thrusters #13-#20 are 25 N

#1-#2 used for positive pitch
#3-#4 used for negative pitch
#5-#6 used for positive roll
#7-#8 used for negative roll
#9-#10 used for positive yaw
#11-#12 used for negative yaw
#13-#20 used for the translation maneuver, and for attitude control(in addition to the other 12 thrusters) during aerobraking.

Thrusters #13-#20 are angled as shown so that the expelled mass does not impinge upon the aerobraking shield.

Placement overview:
The thrusters are placed in those perspective places because of the changing C.M. of the vehicle. The C.M. during the cruise mode is located "under" thrusters #3, #9, #11 and #1 (not shown, on opposite side of vehicle). After the initial ΔV to begin aerobraking the C.M. moves to be positioned "under" thrusters #2, #10, #12, and #4 (not shown.) (the C.M is along the longitudnal axes). This thruster arrangement will minimize errors due to the shifting C.M. The thrusters are able to be fired without introducing a linear acceleration.

This odd arrangement was necessary so that the exhausted propellent mass did not impinge on any other part of the s/c.

5.0Satellite Attitude and Articulation Control System

Due to the operating requirements of the scientific instruments, and communication system, a three-axis stabilization control mode was required for normal on-orbit operation. An initial trade study was conducted to determine the optimum control scheme between:

(1) pure reaction jet, and

(2) momentum exchange system

The trade consisted of a comparison of mass(consisting of fuel and components),reliability,accuracy,and power.

	fuel expended	component mass	reliability	acc	power
pure jet	40kg	.15kg/thruster	jet failure catastrophic	high	negligible
momentum exchange	<2kg	7kg	reaction wheels extremely reliable	high	>10w

Based on the established scoring system, the momentum exchange system was chosen. This was due primarily to the smaller fuel expenditure and greater pointing ability of the momentum exchange system.

The satellite will be using three orthoganal reaction wheels for control about the three principle axes Although this will require sensors for reference about the yaw axis, whereas a momentum bias system doesn't, three-axis stabilization using three momentum wheels was chosen because the satellite will be operating for extended periods of time at both low and high orbit. This introduces greater complexity into the design of a momentum bias(single reaction wheel) system, and also very large wheel momentum requirements. The designed system will allow for the required accuracy (0.1 deg in roll/pitch and 0.3 in yaw ,which was established as the maximum error under which all instruments are operational) ,and small momentum requirements. Under this system, the control torque about each axis is proportional to the rate and attitude error.

5.1Reaction Wheel Sizing

The reaction wheels are sized so that the attitude error remains in specified limits, and so that they do not become saturated when subjected to cyclic disturbing torques.

The solar pressure torque is the primary disturbing torque. It is cyclic about the pitch axis, as the satellite rotates about the planet. The solar pressure at Mars is $P=1.9997 \times 10^{-7}$ N/m^2.

The linearized eq of motion about the pitch axis, for a circular orbit, and 6EO including the gravity gradient is $M_{sy}=I_{yy}\ddot{\Theta}+3\omega_0^2(I_{xx}-I_{zz})\Theta-\dot{h}$, where h= angular momentum of the wheel about the pitch axis, M_{sy} is the solar torque.

The solar disturbing torque during the low altitude orbit is

M_{sy}=((56.2m^2)(.4067m)+(6.82m^2)(.4067m))sinω_0t+Ψ(P)k,

$=6.15\times10^{-6}\sin(\omega_0 t+\Psi)$nm, $M_{sx}=M_{sz}=0$nm, k=1.2 , $\alpha=\omega_0 t+\Psi$

Where α is the angle between the axis of rotation of Mars and the radius vector to the satellite, and k is dependent on the mode of reflection of the incident radiation.The torque is due primarily to the two solar panels, antenna, and satellite body. The solar torque is smaller at the high altitude because of the change in the center of mass. It is

$M_{sy}=3.17\times10^{-6}\sin(\omega_0 t+\Psi)$nm, $M_{sx}=M_{sz}=0$ nm

The wheels will be controlled with attitude and rate feedback.The eq of motion of a wheel is $I_w\dot{\Omega}=\dot{h}$

The change in the wheel angular momentum due to the cyclic disturbing torque may be obtained by the approximating eq $M_{sy}=\dot{h}=I_w\dot{\Omega}$, the solution of which is $\Omega=\Omega_n-\frac{M_{sy}}{I_w\omega_0}\cos(\omega_0 t+\psi)$, where Ω_n is the nominal wheel speed, and ω_0 is the angular rate of the orbit. The change in angular momentum of the pitch wheel is Δ H=.007N-M at low altittude and Δ H=.019 N-M at high altitude. It is assumed that the magnitude of the cyclic disturbing torques about the other axes are approximately equal to that about the pitch axis. The angular momentum of the wheels are nominally zero. The maximum wheel speed is designed so that the wheels do not become saturated when subjected to cyclic disturbing torques and also to minimize dumping. Since the cyclic torques about the s/c are very small,the wheels are designed such that their operating range is 2 n-m-sec. This will insure that the wheels are not saturated by cyclic disturbances due to all causes such as solar radiation, the revolving antenna,etc. Several assumptions have to be made in order to get an estimate of the fuel expended during normal operations. The first is

i) Constant disturbance torques about all axes and due to all causes (gravity gradient,tape recorders,etc) are approximately equal to the maximum solar torque at high altitude i.e. $M_{xd}=M_{yd}=M_{zd}=6.15x10^{-6}$N-M

A simulation needs to be done to get a better estimate for the disturbances.

ii) They are approximately constant over orbit life.

5.2Satellite Tank Size

The fuel expended by the satellite can be divided into three main categories i) fuel expended during normal on-orbit operation ii) fuel expended during the transfer orbit iii) fuel expended for station keeping

5.2.1 Normal On-Orbit Operation

The satellite wheels need to be dumped when the wheel momentum has reached 2 n-m-s Assuming a constant disturbance torque as estimated earlier, $M_{yd}=6.15x10^{-6}$N-M, wheel dumping will be required every

$6.15x10^{-6}$N-M (t)=2N-M-S , so $t=3.25x10^{5}$ sec=3.76days

The impulse needed for dumping over a 4 yr period is

$$(3)2NMS(4yr)(365days/yr)/3.76days=2324.2N\text{-}M\text{-}S$$

The fuel used for wheel momentum dumping over 4 years is then

fuel mass=(2324.2)nms/((340s)(9.81m/s^2)(1.75m)=.7kg

5.2.2 Transfer Orbit

The satellite will be spin stabilized during the transfer orbit at a nominal speed of ω_x=45rpm. The required fuel is (the thruster lever arm is 1.75m)

$2(1.75m)(25N)=3655kg\text{-}m^2\alpha$, $\alpha=2.39x10^{-2}$ rad/sec^2

Impulse=(45rpm)(2π)2(25N)/((60sec)(α)=9858.6N-sec

fuel=9858.6N-sec/((340sec)(9.81m/s^2))=2.96kg

The same calculation is made to calculate the fuel required to despin the satellite,except $I_{xx}=3634kg\text{-}m^2$ .The required fuel is 2.93kg

For the transfer orbit $m_f/1098kg=e^{-1370/((340)(9.81))}$, $m_f=728.14kg$

Fuel mass for transfer orbit is 1098kg-728.14kg=369.86kg.

There is an additional 124.2 kg of fuel for station keeping.It was assumed that the fuel required for station keeping was approximately 33.3% of the fuel required for the transfer orbit. More analysis needs to be done to get a better estimate of the fuel required for station keeping.

mission phase	fuel mass expended
spin-up	2.95kg
de-spin	2.93kg
transfer orbit	369.86kg
normal-on-orbit	0.7kg
station keeping	124.2kg
total fuel	500.64kg

Tank size: 500.65kg/2.6=192.6kg Oxydizer mass=192.6(1.6)kg=308.1kg
MMH mass=192.6kg, so
There are two tanks, each with a volume of 308.1/1431 m^3=.22m^3

5.3Satellite Equipment Characteristics and Location

A trade study similiar to the one conducted for the delivery vehicle was done . Based on the accuracy requirements, sensor weight, and mode of operation, the following equipment was selected.

no per s/c	component	mass	power
1	IR sensor-pencil type	.6kg	.65w
3	sun sensors	.24kg	-
1	sun sensor elec	.55kg	.6w
3	IR sensor-static	1.5kg/each	1.4w/each
1	attitude rate assembly (ARA)	10.08kg	35w
1	reaction wheel assembly	7.0kg	10-20w
2	solar array drive	5kg/each	10w/each

The location of all sensors are outlined in the structures section.

Static IR Sensor

Two are designed for use at high altitude(one redundant), one for use at low altitude. These sensors will provide pitch and roll reference.

mass=1.5kg power consumption=1.4w accuracy=.03(high altitude)
=.05(low altitude)

IR Sensor-"Pencil Type"

Used, in addition to the solar aspect angle(measured by the sun sensors), to provide spin axis reference during the transfer orbit.

mass=.6kg powerconsumption=.65w accuracy=.1deg

Atittude Rate Assembly(ARA)

Unit consists of four gyros (one redundant). It is used for attitude and rate measurements during all phases of mission.

mass=10.08kg power consumption=35w

Sun Sensor Assembly

The sun sensor assembly consists of three digital sun sensors(each 120 deg apart) with an individual field of view of 128x120. Dependent on the mode of operation, the sensors may deliver

i) in spin phase

-solar aspect angle,sun presence signal

ii)in three-axis stabilization phase

-yaw reference

mass: 1 optical head =.08kg
electronics =.55kg power consumption=.6w

Solar Array Drive

Consists of two units which are responsible for rotating the panels to keep them oriented normal to incident radiation. Units designed to meet performance needs. Mass and power requirements based on similiar system.

mass =5 kg/drive powerconsumption=10w/drive

5.4Control Modes

5.4.1Seperation Mode

The initial orientation of the satellite is performed by the delivery vehicle. The satellite is placed in a 400km,circular,polar orbit.(see aerobrake section)

5.4.2Mars Acquisition

The satellite uses rate gyro measurements and IR-static sensor measurements for roll/pitch reference. Yaw reference obtained by sun sensors,and ARA. Control thrusters used to correct large attitude errors due to seperation.

5.4.4Normal On-Orbit Mode

The satellite is three axis stabilized.The three reaction wheels provide attitude control to counteract disturbance torques. Control thrusters used for momentum dumping, station keeping. Pitch/Roll reference is obtained from the IR-static sensor, and ARA. Yaw reference is obtained from the sun sensors and attitude rate assembly(ARA). Solar arrays are deployed.

5.4.5Transfer Orbit

After the scientific mission is completed at low orbit, the satellite will be required to transfer to a high altitude orbit(although it will still be in a polar orbit). The spacecraft will be spin stabilized about the roll axis at 45rpm.The IR-"pencil type sensor",ARA, and sun sensors are used during this phase. Apogee burn and maneuver will be determined autonomously aboard the satellite using on-board computer.

5.4.6 Mars Acquisition/ Normal On-Orbit Operation at High Altitude

These are similiar to those performed at low altitude.The satellite will now use the IR-sensor designed for use at high altitude for roll/pitch reference.

5.4.7 Stationkeeping

It is assumed the Mars station can track the satellite and will be able to command the s/c to perform the needed maneuvers.

5.5 Satellite Thruster Location and Characteristics

There will be 2 thrusters needed for spin-up and 2 needed for despin. These need to be designed so that the rate of angular acceleration is not too low. The thrusters are designed to spin the satellite up to 45 rpm in 3.5min. α=45rpm(2π)(1min/60sec)/((3.5min)(60s/min))=.0224rad/sec^2

$$2(F)(1.75m)=3634kgm^2\alpha\ , \ F=23.3N.$$

There will be 4 25N thrusters for spin/despin and orbit keeping.

The positive /negative yaw, pitch, and roll thrusters, are designed to allow for momentum wheel dumping, and attitude changes when necessary. The NASA STD 5N thruster was chosen due to their great reliability, and versatility. Additionally, the minimum achievable impulse time will allow accuracy requirements to be maintained.

The satellite has 16 thrusters . 4-25N thrusters, and 12 NASA STD 5N thrusters . See Fig 2. for thruster layout.

6.0 Satellite AACS Interaction with Other Subsystems

Satellite aacs interaction with:

i)C^3-primary interaction is concerned with antenna pointing, command. Antenna is able to point independent of the satellite.(see c^3 section).The aacs is linked up to the computer provided in the C^3 subsytem. Total attitude error must be less than .4 deg.

ii)Structure-satellite aacs interaction similiar to that noted under structure-aacs interaction for the delivery vehicle.(see structure section)

iii)PPS-all power for aacs is provided by this system as described. See component descriptions for total aacs power needs.

iv)Science-interaction concerned with the operation, and required

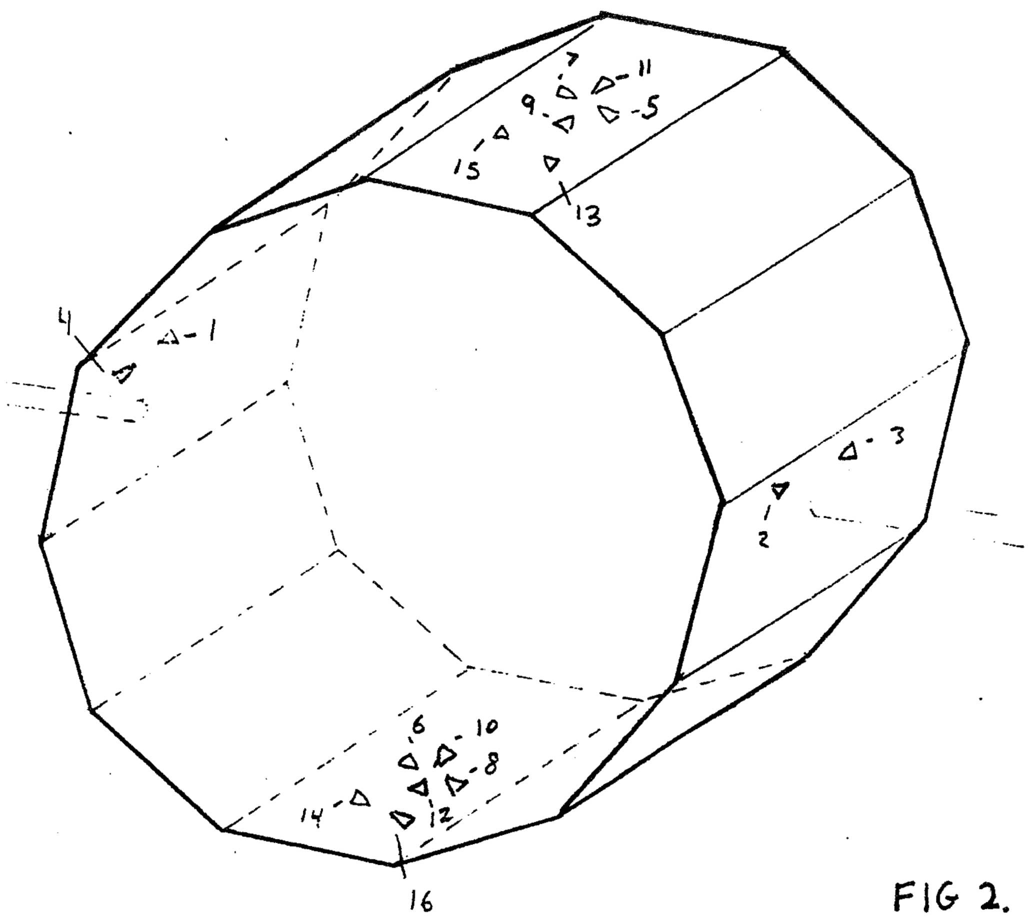

FIG 2.

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Thrusters #1-#12 are NASA STD 5 N
Thrusters #13-#16 are 25 N
#1-#2 used for positive pitch
#3-#4 used for negative pitch
#5-#6 used for positive roll
#7-#8 used for negative roll
#9-#10 used for positive yaw
#11-#12 used for negative yaw
#13-#14 used for spin-up
#15-#16 used for despin

accuracy of the science instruments. Mode of stabilization was dependent on this interaction.See science section for instrument operational requirements

7.0Scanning and Pointing Requirements Implementation

see C^3 section

8.0Critical Areas and Unsolved Problems

-The feasibility of using flaps for control during aerobraking must be studied in more detail.

-The relative stability of the delivery vehicle while it is in the atmosphere was hard to determine. The approximate magnitude of the disturbing torques due to the atmosphere are unknown.

-The magnitude of the disturbing torques(due to causes other than solar) on the satellite are assumed to be approximately equal to the max solar radiation torque. A simulation needs to be done to bound this in a more systematic manner.

-The v needed for station keeping was unknown.

-The placement of the thrusters so that the expelled mass does not impinge on any part of the vehicle is critical.

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ACME

COMMUNICATIONS, COMMAND, AND CONTROL

by CURTIS ZIMMERMAN

[redacted]

REQUIREMENTS

The Communications, Command, and Control (C^3) requirements with respect to the Martian mission fall into four general categories -

1. Uplink - The spacecraft must be able to receive information from Earth or Mars. This information will be in the form of control commands for spacecraft maneuvers, or power switching to seperate subsystems. Also, the spacecraft will receive telemetry from the aircraft operating on the surface of Mars.

2. Downlink - The spacecraft will send information to Earth consisting of data from the onboard science instruments , and overall spacecraft health. Any telemetry or science data received from the aircraft will be relayed to the Martian base.

3. Onboard - The onboard requirements of C^3 include collecting telemetry and relaying commands to the individual subsystems, power switching to subsystems, and data storage. All power required must be obtained from the power and propulsion subsystem.

4. Overall - The C^3 subsystem may have many feasible scenarios. The overall requirements dictate that the C^3 system which proves to be reliable and accurate while minimizing weight, power, size, and cost shall be chosen.

METHOD OF ATTACK - OVERVIEW

The method of attack for C^3 follows four basic steps. First, the entire mission is seperated into operational conditions called mission phases. Communication link objectives and constraints are decided upon for each of these mission phases. Second, a list of communication link options is introduced and a best fit is selected from these options using a trade study weighting scheme. Once the best communications link for each mission phase has been selected, interactions with the other subsystems establishes the input variables for the third step - component sizing. The fourth and final step takes the optimized components and places them on the spacecraft. The specific function of each component is evaluated, and problem areas are analysed. If a specific problem cannot be solved, operational objectives may need to be altered - the last resort being a change in the mission scenario.

MISSION PHASES

The major source of input to derive options for communications link design comes from interaction with the mission planning subsystem. This input can be split up into seperate operational conditions, or mission phases, where each mission phase has its own operational objectives and constraints. Our particular mission has five distinct mission phases.

1. Earth orbit - The C^3 subsystem should provide a link between the spacecraft and Earth for low rate telemetry and routine control commands.

2. En route - The C^3 subsystem should provide a link between the spacecraft and Earth for low rate telemetry and routine control commands.

3. Aerobraking - Although aerobraking will be designed to follow a pre-programmed control sequence, the spacecraft will still need to communicate with Earth or Mars in case of emergency. This communication may occur at any time except during the actual braking in the atmosphere.

4. Mars low polar orbit - C^3 should provide a link (directly, or via Mars base) to Earth for scientific data, as well as maintaining routine command control. Any data gathered while the spacecraft is eclipsed from its link will be stored with magnetic tape devices and relayed at an available time

5. Mars polar synchronous orbit - During this final phase of the mission,

communication between the aircraft and the martian base will be maintained with the satellite bus providing the link. Polar orbit will set constraints on the transmission time (which will be furthur evaluated once the antenna has been sized). Control commands will be maintained in the event of any orbit perturbations.

Once the operational objectives and constraints have been determined, a trade study will determine the optimum communication link design. The following study looks at the transmission link options for the more crucial mission phases. (5=good, 3=neutral, 0=bad)

C^3 TRANSMISSION LINK DESIGN / TRADE STUDY

	Feasibility	Flexibility	Reqs. met	Total
Aerobraking				
high gain/Mars	1	1	3	5
high gain/Earth	0	4	5	9
low gain/Mars	5	3	5	13
Low polar/Mars				
high gain/Mars	4	3	0	7
high gain/Earth	5	2	5	12
low gain/Mars	2	4	0	6
Polar sync/Mars				
high gain/Mars	2	1	3	6
high gain/Earth	0	1	3	4
low gain/Mars	5	3	5	13

Trade study results show that a Mars low gain link should be used for the aerobraking and Mars polar synchronous orbits. A high gain antenna directed to Earth will be used for Mars low polar orbit.

ANTENNA SIZING ROUTINE

Input to size components comes mostly from the science subsystem in the form of data rates. The initial data rate given was 1500 bits per second minimum; however, to make C^3 adaptable to the possible inclusion of a video imaging system, a greatly increased data rate capability was assumed. A high gain antenna was sized using a data rate of 120,000 bits per second (standard magnitude for imaging), and a transmission distance of 2.5 A.U. (maximum distance from Earth to Mars). Other important parameters are given below.

PARAMETERS

B -	bit rate	W_t -	transmitter weight
W -	bandwidth	W_a -	antenna weight
P_r -	received power	W_p -	battery weight
P_t -	transmitted power	R_p -	power weight factor
SNR -	signal to noise ratio	R_a -	antenna weight factor
k -	Boltzmann's constant	r_r -	redundancy factor
dt -	s/c antenna diam.	z -	efficiency
dr -	ground antenna diam.	f -	frequency
D -	transmission distance	λ -	wavelength
T -	receiver noise temp.	C -	speed of light

EQUATIONS

Given the following : B=120,000 Hz, SNR=20, k=1.38 E-23 J/K, dt=64 m, D=2.5 A.U. (3..74 E11 meters), R_a=2.45 kg/m^2, R_p=.227 kg/watt, r_r=4, C=3.0 E8 m/s, z=.55

$B = W \log_2(SNR + 1)$	output W
$SNR = P_r/kTW$	output P_r
$P_t(dt)^2 = P_r(4CD/fzdr\pi)$	output curve of P_t vs. dt

$\Sigma W = W_t + W_a + W_p$ 3.

$W_t = r_r(R_p)(P_r)$	output transmitter weight
$W_a = (\pi dt^2/4)(R_a)$	output high gain antenna weight
$W_p = .082P_t$	output battery weight required
THETA(3dB) = 75 /dt	output beamwidth vs. antenna diam.

Using the previous equations, a graph (fig 1.) is plotted for bulk power vs. antenna diameter, and bulk weight vs. antenna diameter. This plot clearly illustrates an optimum antenna diameter at 3 meters. This size fits well within the limit imposed by the structures subsystem. The power requirement of 50 watts for this size antenna was given to the power and propulsion subsystem and was considered acceptable.

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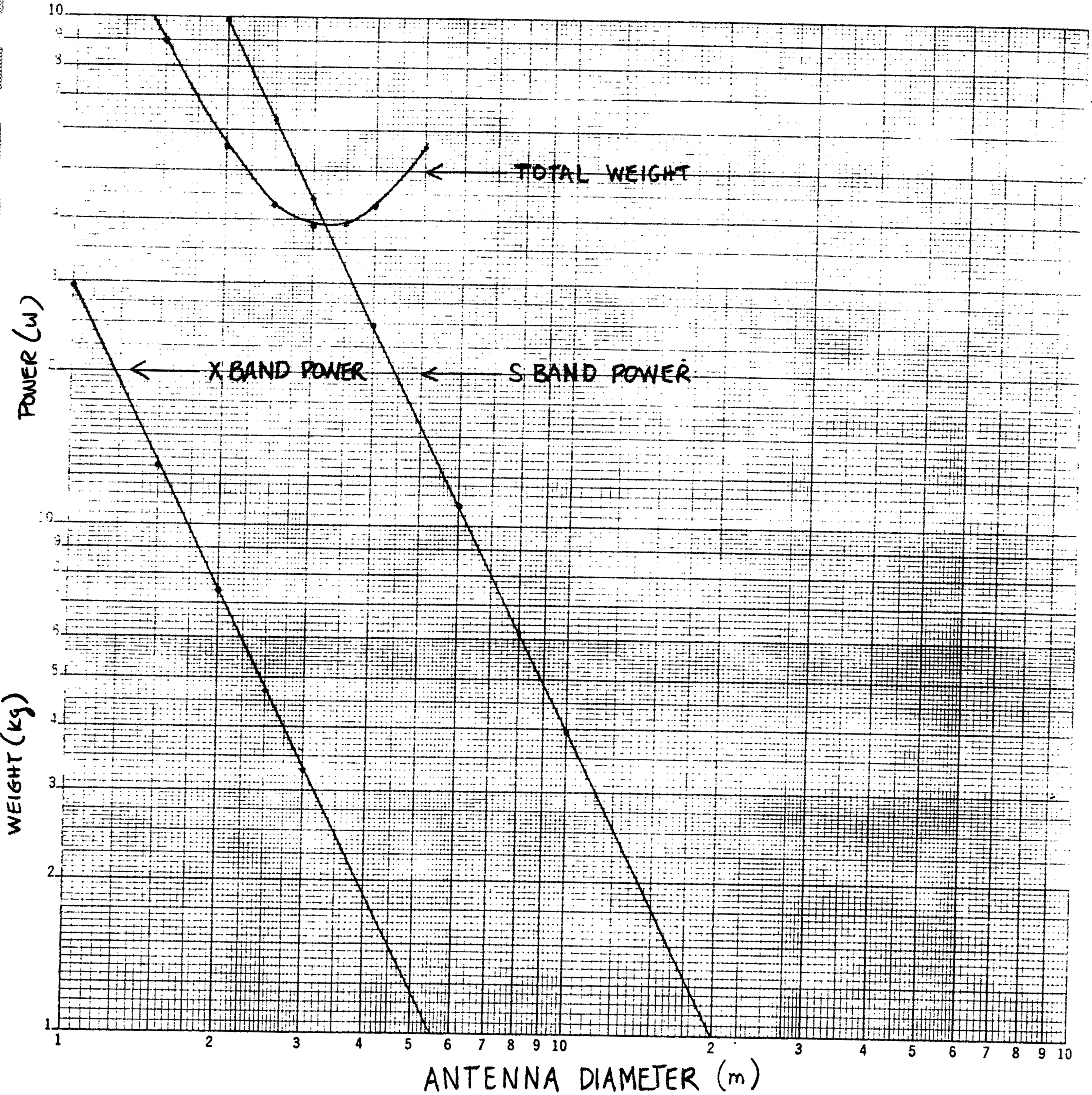
WEIGHT AND POWER vs. ANTENNA DIAMETER



Logarithmic, 2 x 2 Cycles

ANTENNA POINTING

Pointing accuracy is necessary in all mission phases involving high gain data transmission. The antenna will be configured such that it has two degrees of rotational freedom and will be able to orient itself using attitude and control sensors. The degree of accuracy of the sensors can be calculated using the beamwidth equation found in the antenna sizing section. For a 3 meter antenna using a wavelength of .0357 meters, the beamwidth is calculated to be .9 degrees. The attitude and control sensors will need to be accurate to THETA/2 degrees, or .45 degrees.

TRANSMISSION SCHEDULE

During the final mission phase, the spacecraft bus must provide a communication link between Mars base and the aircraft. The desired beamwidth for global coverage was calculated to be 19.4 degrees using the THETA equation and orbit geometry values. To project this beamwidth, a low gain horn antenna with a diameter of .14 meters will be used. The projected transmission "footprint" (shown in fig 2.) has a radius of 2075 miles (compared to the Martian radius of 2105 miles). Because the footprint cannot cover horizon to horizon, there will be two circumscribed areas where transmission cannot occur (shown in fig 3.). The radius of these areas is 354 miles. These non-transmission regions are further enlarged due to the footprint and the Mars base describing independent

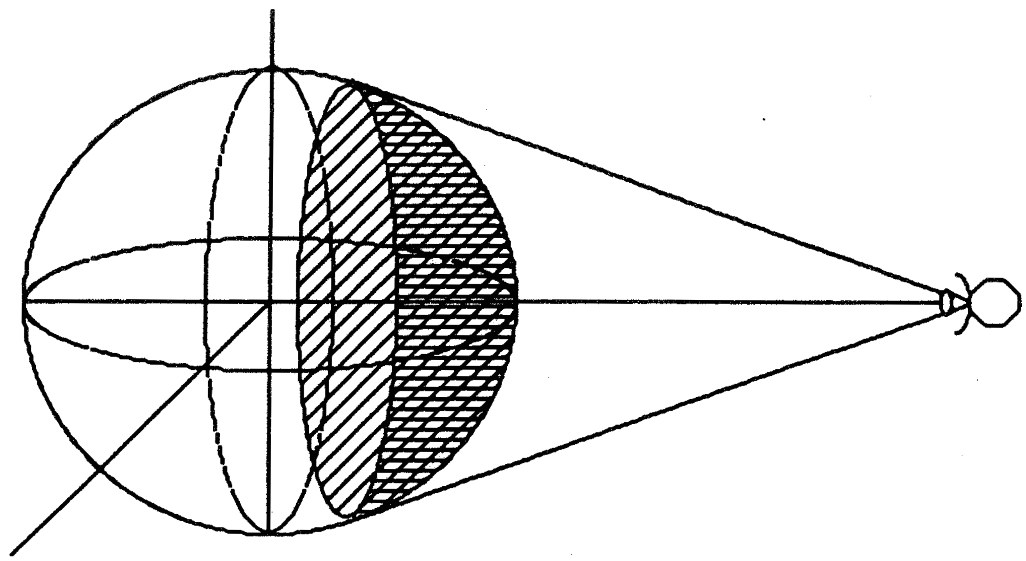


FIG 2. "FOOTPRINT" OF A LOW GAIN ANTENNA ON THE MARTIAN SURFACE

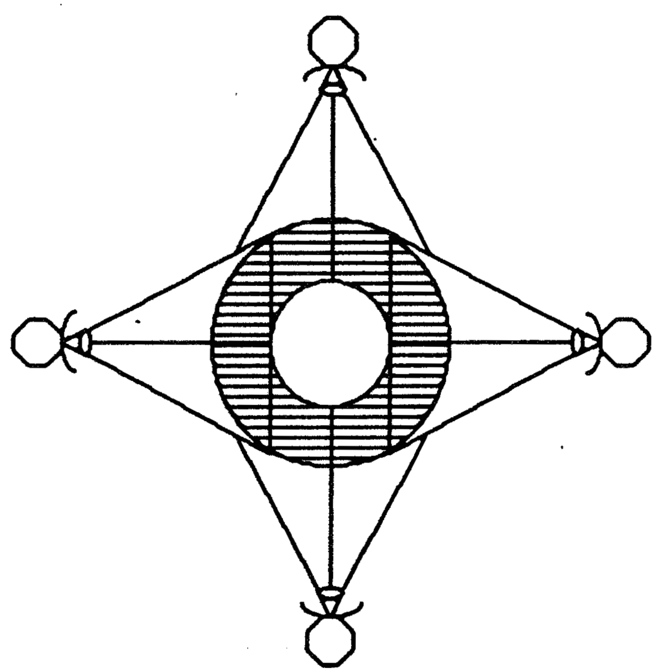


FIG 3. TRANSMISSION CAPABILITY OF LOW GAIN ANTENNA IN POLAR ORBIT. BLACKOUT REGION IS CENTRAL AREA.

circular paths in two orthogonal planes.[2] When the Mars base has moved past the edge of the small non-transmission zone, it still is not within the area covered by the footprint. Total transmission blackout time can be estimated using an average velocity for the Mars base, and an average rotation rate for the footprint. The applicable geometry is shown in fig 4. The total blackout time was calculated to be approximately 4 hours out of every twelve - the airplane duration of 8 hours was accommodated.

Another idiosyncracy of the polar orbit besides transmission blackout is the variable transmission schedule throughout the Martian year (shown in fig 5.). Using a Martian year of 687 days, and the geometry shown in figure 5 (which assumes sunrise at 6:00 AM, and sunset at 6:00 PM), the blackout time was calculated to occur .035 hours earlier each day. Assuming the aerobraking insertion orbit to occur exactly orthogonal to the Mars-Sun line, an aerobraking period of 54 days, and a low orbit period of 21 days, the transmission blackout will begin at 8:22 and end at 12:22 on the first day of the final mission phase. The aircraft can communicate with the Martian base from 12:22 PM to 8:22 PM, and from 12:22 AM to 8:22 AM. At any time during the year, the maximum flight time which could occur in the dark is 4 hours.

PROBLEM AREAS

The major problems to be dealt with concern the dynamics of the polar orbit and their effect on communications. First of all, the placement of the satellite bus from low polar orbit to polar sync orbit is crucial. The

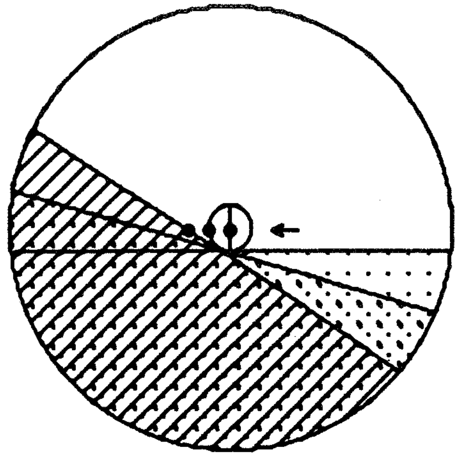


FIG 4. EXTENDED BLACKOUT TIME - PROJECTED VIEW

- mars base in center of blackout region

- mars base at edge of blackout region

- mars base at edge of transmission footprint

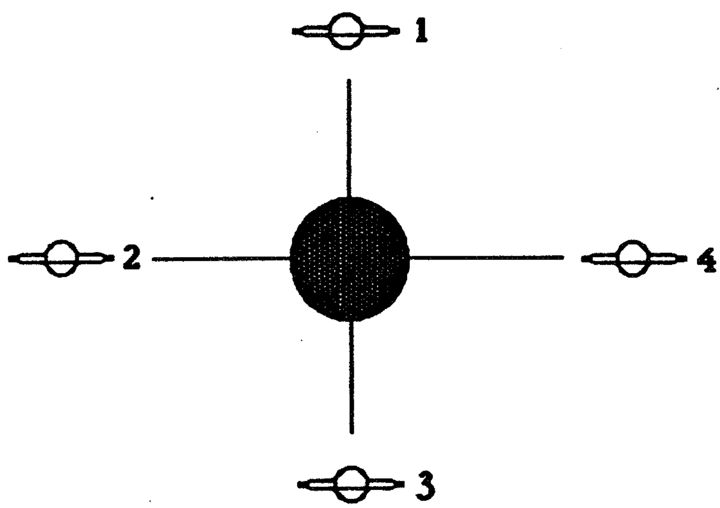


FIG 5. EFFECT OF MARTIAN HELIOCENTRIC POSITION ON TRANSMISSION TIME. 1&3 HAVE BLACKOUT FROM 10:00 TO 2:00. 2&4 HAVE BLACKOUT FROM 4:00 TO 8:00

conjunction of the central point of the footprint and the Martian base must occur in the plane of the orbit for maximum coverage. Also, an assumption was made that the satellite bus maintainted a constant velocity during the final mission phase. The effects of orbit perturbations (possibly due to oblateness) and the effect on communications will have to be considered. If this presents too much of a problem, a less stringent communication requirement may have to be adopted.

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ACME Science Subsystem (SS)

Designed by:

Mike Scheller

Introduction

This section of the ACME final design report contains the science subsystem requirements, method of attack, and final design.

Method of Attack-Overview

The first step in the method of attack is receiving and identifying the requirements for the subsystem after which, research is initiated to identify potential science instruments that would be of use to the mission. Through many hours of research in the libraries, a general list of instruments is organized. Instruments that perform the same experiments are compared to determine which instruments will best fulfill the requirements of the RFP. After the final set of instruments is decided upon, their vital parameters are given to each subsystem that interacts with the science subsystem. If these parameters are not suitable for the other subsystem plans, more research and/or trade studies will be done.

Requirements

Specific long and short term goals of the science mission were also recieved from M. Lembeck. These are listed below.

Long term goals

+ Determine the origin, evolution and present state of the solar system.
+ To better understand the Earth through comparative planet studies.
+ To understand the relationship between the chemical and physical evolution of the solar system and the appearance of life within it.

Short term goals

+ Determine the elemental and mineralogical characteristics of the Martian surface on a global basis.
+ Determine the distribution, abundance, sources and concentrations of volatile materials and dust.
+ Define the global gravitational field.
+ Measure the global topography.
+ Explore the atmospheric structure and circulation in detail.
+ Establish the nature of the Martian magnetic field.

Assumptions

The following assumptions are used in analyzing the science subsystem. They are invoked at different times during the analysis but are listed in entirety here.

+ science instruments will not be used during Earth-Mars transfer.

+ instruments may very on each of the four data buses
+ approx.imately one month will be available for low orbit experiments
+ mars base will be able to gather atmospheric data using weather balloons
+ data from MGCO will be used to supplement data gathered by ACME SS
+ instruments may be modified to work at high altitude

Research methods

With the requirements properly identified, it is now necessary to determine ways to locate science instruments that may be used to fulfill the requirements. Two research options became apparent: research past missions or research the action required of the instrument.

To research past missions it is necessary to identify missions that may be useful. The Viking craft , Mariner Mark 4 and the Mars Geoscience Climatology Observer (MGCO) are best suited for research since they are fairly recent missions and share science objectives similar to ACME SS. These craft are referenced througth the library's computer system, STAR, IA abstracts and any other source that may appear helpful. The instruments used on these craft are then used as a basis for instruments used on ACME SS.

Researching instrument actions or uses involves using library computer terminals in their subject search mode. Subjects such as 'gravity measurement' and 'doppler shift measurement' are used to identify sources of information.

Research results

Researching past and future missions gives the most information although alot of further research was done on specific instruments themselves. Instrument use searches results in alot of information about Earth-borne experiments. A list of components can be compiled from these searches that will lead to a final science package. The components selected can be found in table SCI-1. These parameters contribute to the requirements of the other ACME subsystems.

Subsystem interaction

The structure subsystem depends directly on the mass, size and placement of the instruments. AACS, power and aerobrake subsystems also depend indirectly on these parameters. Thermal considerations affect the structures subsystem. Mission planning recieves the modes of operation and their times. Pointing and scanning motion requirements directly affect the AACS subsystem. The data rates for the science package are used by the C^3 subsystem to calculate transmission rate information.

Selection considerations

For comparisons of instruments used for similar experiments see table SCI-2a through 2c.

It was decided that an imaging camera would be a luxury that would not be needed if the uv-vis-ir mapper was used. Allowances are made with C3 to accomodate a camera at a later date if needed.

Table SCI-1 Initial Component Parameters

component	method/use	status	mass	power	data	size/placement
He vector magn.[1,6]	measure magnetic fields	M	3.46	7.3	150	10^dx15/ boom
geiger tube tele.[8]	measure energy particles	M	N/A	N/A	N/A	req. 2 d.o.f.
cosmic ray tele.[8]	measure hi energy particles	M	N/A	.625	N/A	2 d.o.f.
trapped rad. detc[8]	maps electron energy spect.	M	3.5	.279	N/A	N/A
UV photometer [8]	detects H and He	M	N/A	.6	N/A	10x10x12
nephelometer [8]	measures cloud cover	M	N/A	N/A	N/A	N/A
IR reflect. mapp.[8]	det. mineral phase and com.	M	16.2	8	N/A	30x40x40/ 2 dof
gas chroma. [8]	measures amt of noble gas	M	N/A	N/A	N/A	N/A
radar altimeter[1]	measures surface topo.	OS	7	17	625	29x29x19
altimeter dish	-	OS	-	-	-	1 m diameter
radio science [1]	measure gravity w/ doppler	M	-	-	-	use radio trans.
gamma ray spect.[8]	measure elemental comp	M	11	2	2000	40x40x30/ 2dof
ion mass spect. [8]	maps comp of ionosphere	M	2	.08	N/A	2 dof
imaging camera[3]	visual study of mars	M	28	10	2Mbs	2dof 30x30x40
UV spectrometer[1]	map ozone	M	11.3	12	200	54x28x40
gradiometer[5]	map gravity field	UD	N/A	N/A	N/A	76x76x20
uv-vis-ir spect[2]	map elem./min makeup	M	25*	20*	2000	31x40x40
PMIRR [4]	map atmos/map volatile matl	UD	25.7	27.1	150	81x33x24 70x40x63

The only field of view characteristics found were
PMIRR- 1x1degrees
uv-vis-ir mapper- 45x45 degrees

All instruments face Mars with underlined dimension
For specific locations of instruments see structures subsystem
* number assumed from similar instruments

mass is measured in kg

power is measured in watts

data rates are measured in bits/sec unless otherwise noted

all sizes are in cm

M = modified instrument
UD= under development
OS = off-the-shelf

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Table SCI-2A

Evaluation of atmospheric mapping instruments	
package A	Package B
geiger tube telescope cosmic ray telescope trapped radiation detector uv photometer nephelometer ion mass spectrometer radio science gamma ray spectrometer	-gamma ray spectrometer Pressure modulated infrared radiometer (PMIRR)
Advantages	Advantages
+ off-the-sheif technology + proven reliable on previous missions	+ low mass + less power required + less space required + also maps volatile materials
Disadvantages	Disadvantages
+ technology is not up to date + high mass + high power consumption + large dimensions + cost is not efficient + no data rates available	+ technology is not perfected yet + not prove/tested
Statement: Package B is chosen since it meets the reqirements of low mass, performs the necessary experiments and simplifies the inertia tensor of the instrument bus.	

Evaluation of the volatile material mapper
Package A -PMIRR and IR mapper
Statement: Since the PMIRR is already being used and no other instruments of its caliber are found, it will be used on the bus. The uv-vis-ir mapper will also contribute to the data on volatile material.

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Table SCI-2B

Evaluation of global topography measuring instruments	
Package A- radar altimeter	
Advantages	**Disadvantages**
+ low mass + proven effective + low power required	+ only instrument found to achieve required data + new technology needed
Statement: The radar altimeter will be used but since there was no other instrument found further research should be done.	

Evaluation of elemental and mineralogical mapping instruments	
Package A	**Package B**
gamma ray spectrometer uv spectrometer ir spectral mapper	gamma ray spectrometer uv-vis-ir mapping reflectance spectrometer
Advantages	**Advantages**
+ proven technology	+ conserve space + added benefit of a visual spectrometer + low mass
Disadvantages	**Disadvantages**
+ uses alot of space + excess mass and power	+ no proof of technological accomplishments can be found
Statement: The gamma ray spectrometer and uv-vis-ir spectrometer was chosen because the gamma ray spectrometer is used in the atmosphere experiment which reduces the number of instruments needed by one. The uv-vis-ir spectrometer combines three instruments into one which also decreases the number needed.	

Table SCI-2C

Evaluation of gravitational field mapping instruments	
Package A- gradiometer	Package B-radio science
Advantages	**Advantages**
+ recent technology + higher degree of accuracy + experiment is self contained	+ proven technology + radio is needed anyway therefore there is no extra mass
Disadvantages	**Disadvantages**
+ must be spun at 240rpm + not tested in space + relatively large	+ polar orbit is difficult for transmission
Statement: Since radio science experiments have been done sucessfully in the past and it is proposed to have a link with Earth anyway, it is suggested that radio science experiments are conducted. Gradiometeric experiments may be used if later study proves that they would be efficient.	

Evaluation of the magnetic field instruments	
Package A - Helium vector magnetometer	
Advantages	**Disadvantages**
+ reasonable mass + tested in space-reliable + low power required	+ only instruments found - could be better instrument + must be boom-mounted-- effects stablity
Statement: The helium vector magnetometer will be used on the bus unless at a future date a better instrument is developed which has better characteristics of the HVM.	

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Modes of operation

As consequence of data output rate , the science instruments will be operated in three (3) modes. While one mode is running, the others will be downloading data. A direct result of this will be the use of tape recorders to retain data. Information will be downloaded to the recorder which will in turn download the data to Earth at a convienent time.

Mode 1 will consist of the PMIRR and uv-vis-ir mapper. This mode will operate only during the time that the landing site needs to be investigated before the payload is sent to the surface. Starting time (date) for this mode will be determined at a later date as will the total time of operation.

Mode 2 will consist of the uv-vis-ir mapper, radar altimeter and magnetometer. These instruments were chosen so the visual mapper and radar altimeter maps will coincide. The magnetometer will run constantly since it has a low data rate.

Mode 3 will consist of the magnetometer , PMIRR and the gamma ray spectroscope. The PMIRR and gamma ray spectrosope were chosen to operate together so that the volatile material map and the elemental /mineralogical map will coincide. For a list of mode requirements for power see table SCI-3.

The radio science experiment will take place whenever a transmission is made to Earth.

Testing and Calibration

All instruments must be bench tested individually and then

Table SCI-3
Modes of Operation

Mode	power req (w)	Data rate(bps)	record time(hr)	orbits
Mode 1	47.1	2650	10.48	5
Mode 2	44.3	2300	12.08	6
Mode 3	36.4	3275	8.48	4

Time to record data= capacity of tape/data output rate

Orbit to fill tape = time to fill tape /1.96hr/orbit

Time to downlink full tape= data amount/transm. rate

Tape recorder can store 1.7x10^8 bits of data and
can transmit at 29.4kbs

integrated into one science package. This package is then integrated with C^3 and AACS to determine if all modes, movements and commands are carried out properly. Instruments will be calibrated on Earth before departure and calibration checked when orbit is achieved.

Problem areas

Problem areas have arisen in most parts of this design. The areas that need further study are listed below with a brief explanation.

1. Payload departure from craft.- Since an imaging camera will not be used, the uv-vis-ir mapper will determine the suitability of the landing zone. This instrument has a 45x45 degree f.o.v. This will permit the instrument to see the mars base approximately 400 km before it is directly above the base. If the payload is released at this precise moment it will over shoot the base by approximately 984 km. This was calculated assuming free fall and no control forces using the following equations.

$$\text{time to free fall} = (\text{altitude} \times 2/\text{gravity})^2$$

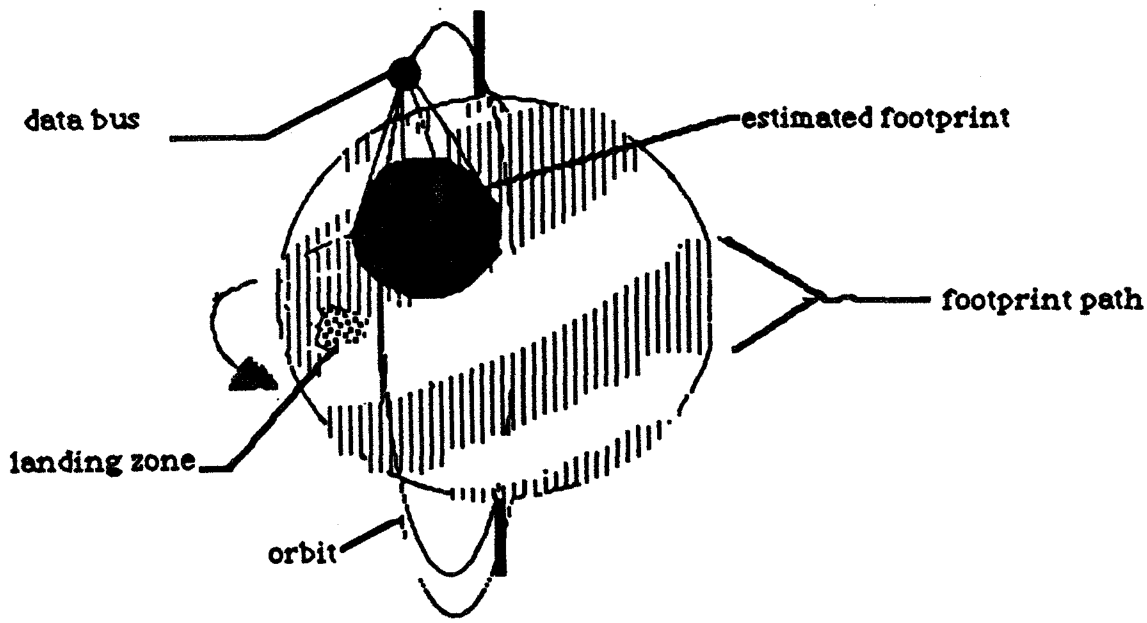
$$\text{distance payload travels} = \text{time to fall} \times 2.88\text{km/s}$$

To correct this overshoot retro-rockets and directional parachutes can be used or the payload can be released on the next orbit. This however results in an under-shoot to the west of the base.

2. Instruments-
   a. Data rates for instruments were very hard to locate and may not be accurate. Instrument sensitivities and accuracies were generally not available. Therefore more research is needed.

b. Time at low Mars orbit is minimal. Tests should be run to determine if science instruments will be accurate enough at high orbit to gather more information.

c. The requirements of global coverage need more study to determine if in fact, the entire planet will be covered.(see fig SCI-4) The path of the footprint will be a type of helix. The total area covered by this helical band is very important but is considered in the next design phase.



Final Design

ACME SS is composed of a pressure modulated infrared radiometer, a magnetometer, a gamma ray spectrometer, uv-vis-ir mapping reflectance spectrometer, radar altimeter, and a radio science experiment. Two tape recorders are used to buffer data being sent to earth. Total system has a mass of approximately 77.2 kg and requires 91.4 watts of power. Three modes of operation will be used with data rates of 2650, 2300, and 3275 bits per second each.

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ARES PROJECT

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VI. Chapter Six (CCC)

VII. Chapter Seven (SCIN)

The following is a list of the system requirements as presented in the document entitled "Request for Proposal and Preliminary Design of a Manned Mars Aircraft Space Delivery System." (RFP).

System Requirements

1) Develop A conceptual design for the spacecraft delivery system required to deliver a manned aircraft to the Martian surface in the first decade of the next century.

2) The spacecraft will consist of two primary components: the payload delivery system and an orbiting instrument bus.

3) The following subsystems are identified for the purposes of system integration:

a	Aerobrake	(AERO)
b	Structure	(STRC)
c	Power and Propulsion	(PPS)
d	Attitude and Articulation Control	(AACS)
e	Command and Data Control	(CCC)
f	Science and Radio Relay Instrumentation	(SCIN)
g	Mission Management, Planning and Costing	(MMPC)

4) Nothing in the Spacecraft's design should preclude it from performing several possible missions, carrying vastly different payloads to different destinations.

5) The design should use off-the- shelf hardware where available.

6) The design should not use materials or techniques expecxted to be available after 1998.

7) The spacecraft should have a design lifetime of at least four years.

8) The vehicle will use the latest advances in artificial intelligence where applicable to enhance mission reliability and reduce mission costs.

9) The design will stress simplicity, reliability, minimum mass and low cost.

In accordance with requirement 3, the subsystems present their requirements in the following chapters. Final designs are presented in these sections.

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MMPC SUBSYSTEM

The following requirements are specific to the MMPC subsystem.

- Select target
- Compute ΔV requirements
- Describe trajectory
- Design the spacecraft to carry out the mission
- Build the components of the four delivery systems
- Integrate the spacecraft
- Test the spacecraft
- Launch the spacecraft
- Provide mission support

In order to fulfill the system and subsystem requirements, three main tasks are identified. Within these tasks, the derived requirements, options, design trades, and methods of attack are presented. Completion of these tasks leads to a final design; this design is also presented.

MAIN MMPC TASKS

Escape From Earth Orbit
Mars Orbit Insertion
Mission Implementation

ESCAPE FROM EARTH ORBIT

This task involves calculating ΔV requirements and determining orbit trajectories. The trajectory that best satisfies the requirements (most notably, the minimum Δv requirement) is chosen for the final design.

METHOD OF ANALYSIS

Four possibilities were considered for the method of escape from Earth orbit. From PPS, two types of propulsion systems were considered feasible for this mission. For each propulsion system, two types of orbit transfers were considered. The resulting four methods are:

Low Thrust Direct Transfer	LTDT
Low Thrust Venus Flyby Maneuver	LTVF
Impulsive Direct Transfer	IDT
Impulsive Venus Flyby Maneuver	IVF

All four types of transfer were analyzed using the patched conic method. This means that thrust arcs occured within the sphere of influence of the respective planets. The ΔV required from the thrust maneuvers is determined from the hyperbolic excess velocity (V_{he}) needed to put the spacecraft on a free-fall heliocentric trajectory to the target planet. The characteristic ΔV and V_{he} needed for the transfer is a function only of the heliocentric transfer geometry.

The MULIMP program was used to determine the local minimum ΔV and V_{he} requirements for the two types of transfers. The ΔV data was used in the analysis of the impulsive systems while the V_{he} data was used to determine characteristics of the low thrust propulsion systems.

ANALYSIS - DIRECT TRANSFER (DT)

The MULIMP program was used with the parking orbit at Earth set at 250 nautical miles (elements of the capture orbit are presented in the section: Mars Orbit Injection). A step search was done to find the first local minimum in the launch window. A total time optimization (TMFIX = 0,0) was then performed for the time of the first minimum and also for the next two synodic periods.

ANALYSIS - VENUS FLYBY MANEUVER (VF)

Since the synodic period is larger for Venus with respect to Earth than it is for Mars with respect to Venus, The opportunities for a VF maneuver are limited by the Earth-Venus geometry. With this in mind, a MULIMP search was performed and the minimum energy transfer from Earth to Venus was found. The date of this launch was used in a time optimizing (TMFIX = 0,0,0) analysis of the VF maneuver (the parking and capture orbits were those specified in the DT). The radius of the gravity assist maneuver was also optimized. The analysis was also performed for the next two synodic periods.

For both cases, the earliest launch date was the least costly in terms of ΔV requirements. The earliest launches were then compared to determine which of the two would become the final design choice.

MULIMP OUTPUT FOR ΔV ANALYSIS OF DT AND VF TRANSFERS

	Launch Date	ΔV Total (km/sec)	ΔV Escape (km/sec)	V_{he} Earth (km/sec)	V_{he} Mars (km/sec)
DT	June 8, 2003	4.456	3.563	2.970	2.700
VF	June 5, 2004	7.310	4.122	4.653	6.190

FINAL DESIGN

The VF maneuver requires a 15.7% increase in ΔV_{escape} and a 64% increase in ΔV_{total} as compared to the DT. Also, the geometry of the VF maneuver is such that the spacecraft would arrive at Mars just before perihelion passage of the planet. Since at this time Mars experiences a major global dust storm, the spacecraft would be required to wait in orbit for a few months until the aerobrake maneuver could be initiated.[1] Because of the undesirable attributes of the VF maneuver, the DT was chosen.

DT CHARACTERISTICS-FINAL DESIGN

Launch Date	June 8, 2003
Arrival Date	Dec 29, 2003
Transfer Time	204 days
ΔV Escape	3.563 km/sec
V_{he} (Earth)	2.970 km/sec
V_{he} (Mars)	2.700 km/sec
Distance to Earth at Arrival	1.0865 a.u.
Orbit Elements of Transfer Ellipse:	
Semimajor Axis	1.26 a.u.
Eccentricity	0.1946
Inclination	0.00 deg
Longitude of Ascending Node	undefined
Longitude of Periapse	-105.9 deg
Time After Periapse Passage (at launch)	2.14 days

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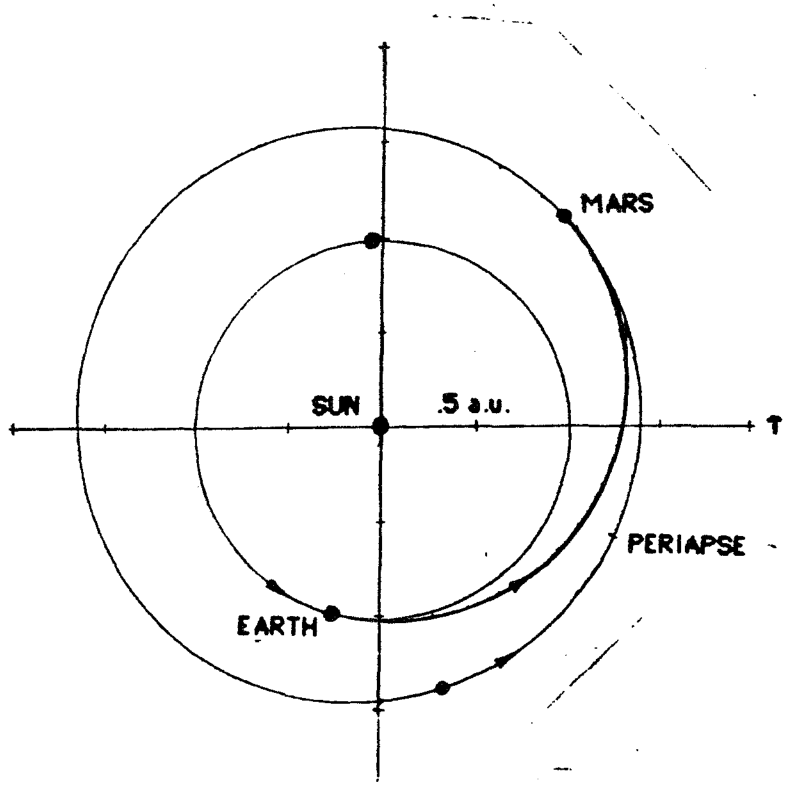


IDT TRANSFER ELLIPSE

As shown above, the spacecraft encounters Mars well past perihelion passage. The trajectory also exibits the advantage of having a short communication distance to Earth (18 min. round trip signal time) at time of arrival.

ANALYSIS-LOW THRUST

Before any analysis was done on the low thrust trajectory, an extensive literature search was conducted. The most notable findings are listed as references 2 and 4. Reference 4 presents the state differential equations for optimal guidance on a low thrust trajectory. Solving these equations requires a numerical routine to solve a two point boundary value problem (TPBV). Due to time constraints, using such a routine to calculate a trajectory was not considered. Reference 2 outlines a method of solving for the non-optimal trajectory. Charts are given that plot radius vs. time and time of burn vs. V_{he} (the quantities are non-dimensionalized to make the charts universal). Interpolation of the charts for the exit velocity of the candidate engine and for the V_{he} of the transfer orbit and for all the quantities non-dimensionalized with repect to the initial parking orbit yielded the following results.[2]

$$a = a_0 g_0 = \text{Thrust / Mass} = \text{initial thrust acceleration}$$

$$a_0 t_0 \approx 1$$

$$t = t_0 * 896.253 \text{ sec} = \text{time of thrust}$$

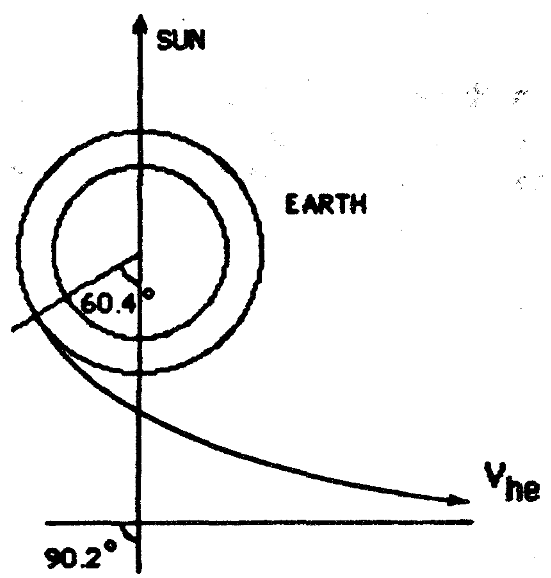
$$g_0 = 8.5166 = \text{local gravity acceleration on parking orbit}$$

These relationships were given to PPS to determine if a propulsion system could be designed such that the thrust provided and the initial mass of the spacecraft would produce a mission of reasonable length.

From PPS it was reported that power requirements make the low thrust option not feasible.

FINAL DESIGN

The choice for method of escape from Earth orbit is the IDT maneuver. The geocentric trajectory is shown below.



Technical Problem Areas

- Not able to solve for optimal low thrust trajectory for comparison with impulsive transfers.
- Had difficulty finding current research on low thrust trajectories.

Analysis of VF trajectories was difficult since MULIMP failed to converge for most launch times other than the three noted.

EFFECTS ON OTHER SUBSYSTEMS
Communication distance affects antenna sizing (CCC)
Transfer geometry affects antenna pointing (AACS)
ΔV requirements affect fuel requirements (PPS)

MARS ORBIT INSERTION

This task involves putting the spacecraft on a trajectory to allow initiation of the aerobrake maneuver.

METHOD OF ANALYSIS

Since no significant data was obtained from the AEROB program, an alternate method of determining fuel and ΔV requirements had to be found. The analysis invoves assuming characteristics of the aerobrake trajectories and computing ΔV at the three burn times. The characteristics of the aerobrake trajectories were chosen such that the AERO subsystem believes the aerobrake will be able to serve its purpose (i.e. circularize the initial orbit).

ANALYSIS-ΔV CALCULATIONS

The assumptions about the aerobrake trajectories are as follows.

Initial Semimajor Axis	2000000 km
Initial Periapse Radius	3600 km
Altitude of Periapse (for breaking)	60 km
Final Orbit Radius	4722 km

The choice of final orbit radius is presented in the section: ANALYSIS-FINAL ORBIT. Three ΔVs were calculated. The first ΔV is at the initial periapse radius and is done to achieve the high energy elliptic orbit. The second burn is at apoapse of the high energy ellipse (and is initiated when it has been determined that the global dust storm has indeed died down) to bring the periapse down sufficiently far into the atmosphere to allow aerobreaking. This burn also allows for a 66 degree change in inclination of the orbit. The circularizing burn occurs at apoapse of the last elliptical trajectory (when apoapse radius is equal to the final orbit radius). The numbers above and the vis-viva equation

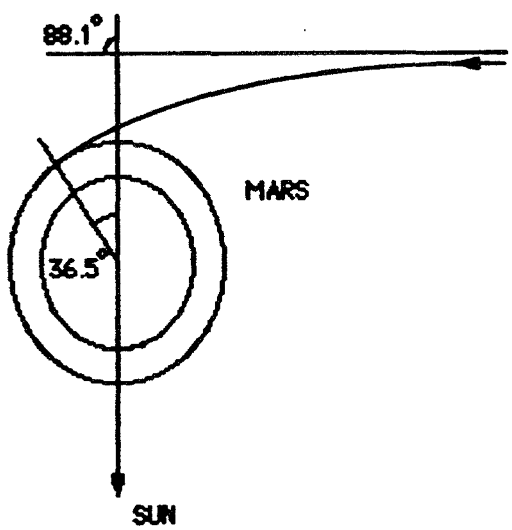
$$V^2 = \mu(2/r - 1/a) \quad (3)$$
$$\mu_{Mars} = 4.305E{+}04 \ kg^3/sec^2$$

are used to determine the velocity vectors before and after each burn. The change in velocity yields the ΔV for each burn.

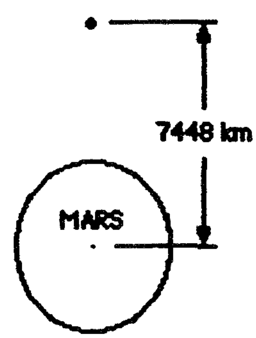
ΔV CALCULATIONS FOR ORBIT INSERTION

ΔV_1	717.890	m/sec
ΔV_2	43.937	m/sec
ΔV_3	247.230	m/sec

For comparison, the ΔV required for orbit insertion to the final circular orbit from hyperbolic approach is 2566.7 m/sec.



MARS ORBIT INSERTION



AIMING RADIUS

The geometry of the orbit insertion is presented above. Note that the circular orbit shown for insertion represents the radius of the periapse and not the orbit trajectory of the high energy ellipse.

ANALYSIS - FINAL ORBIT

The orbiting bus must be placed in an orbit such that the science objectives are met and that the airplane support requirements are met. There is some conflict involved in trying to fulfill the requirements of both with

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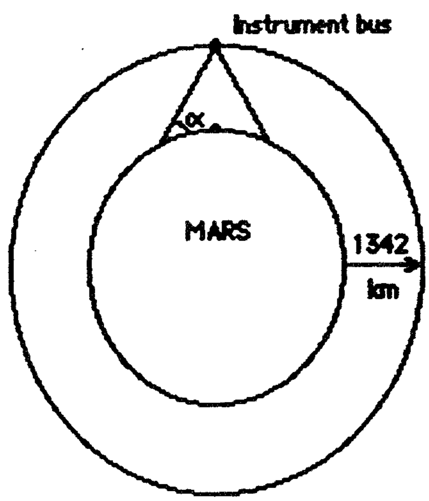
one orbit. From SCIN, the most desirable orbit for global science requirements is a low circular polar orbit. For support, a synchronous orbit over the Martian base would be desirable. Characteristics of both were combined.

A circular orbit with a period equal to some fraction of the period of Mars' rotation (24.623 hours)[4] and inclined at some angle satisfies the requirements of both groups to some degree. The orbit passes over the base with some frequency and is able to image a great portion of the surface of Mars.

Kepler's third law and the vis-viva equation combine to produce two main categories of orbits: low-fast orbits and high-slow orbits. Because the slower orbits require less ΔV of circularization and allow for less complicated control of the scan platform and allow greater communication times with the Martian base, it was decided that attaining the highest orbit possible would be advantageous. From SCIN, an altitude of less than 1500 km was desired for resolution purposes. The lowest integral number of orbits per day that produces an altitude less than 1500 km is 9.



MARS ORBIT
(9 orbits per day)

The angle alpha in the picture above is called the angle of incidence and it is a measure of the amount of area that an instrument can observe from one point in space. Representative numbers for alpha could not be found by SCIN. On succesive orbits, the ground track moves westward 360/9 or 40 degrees. Given the orbit altitude, CCC designed a UHF antenna with a transmission distance of about 1800 km. With simple geometry and noting that the velocity on the orbit is 3.02 km/sec, communication time with the base is

15.7 minutes each flyover.

The inclination of the equator of Mars to the ecliptic is 23.984 degrees. This means that coming in on the ecliptic plane will automatically put the spacecraft on an orbit inclined to the Martian equator. The allowance for the 66 degree inclination change stated earlier allows the instrument bus to be put in a polar orbit. As shown the previous section, this large change in inclination is done with a small increase in fuel requirements. The final orbit has the benifit of providing global coverage.

The final orbit can be positioned to pass directly over the Martian base no matter where the base is located on the surface of the planet. If the base is on the equator, the instrument bus will pass over the base two times per day (once going northward and once southward); if at a pole, nine times per day; and if otherwise, once per day. This orbit would be positioned so that the instrument bus would pass over the Martian base during daylight. The airplane science missions would be run to coincide with the overhead passes.

TECHNICAL PROBLEM AREAS

No numbers were found for critical angle of incidence of the instruments needed to determine ground coverage

Interaction with the aircraft group revealed poorly defined requirements

Oblation effects cause a change in the radius required to achieve the desired orbital period. This difference is not significant at this level of design.

EFFECTS ON OTHER SUBSYSTEMS

ΔV calculations affect fuel requirements (PPS)

Orbit geometry affects antenna sizing, instrument pointing, solar array pointing, data transmission time. (AACS) (CCC)

Orbit frequency affects timing of aircraft science missons.

MISSION IMPLEMENTATION

This task involves outlining the timeline of the mission and performing cost estimates of the mission.

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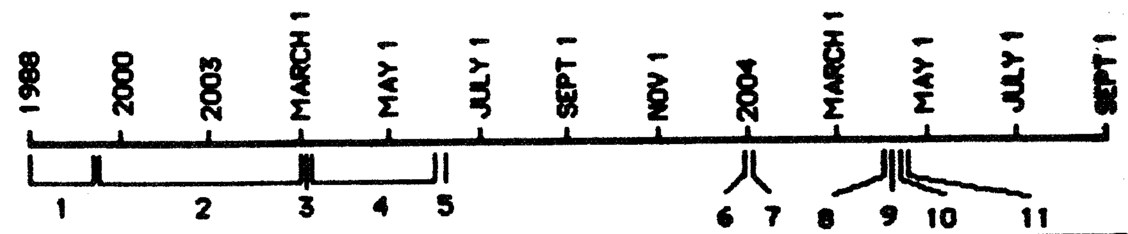
TIMELINE



1) Development phase (detailed analysis, design, testing of subsystems)
2) Constuction and integration (subsystem testing, spacecraft integration and testing)
3) Launch to Space Station
4) Assembly and test on orbit
5) Launch of spacecraft from Earth
6) Arrival at Mars
7) Initiation of aerobrake maneuver
8) Orbit Circularization
9) Determination of landing sight viability
10) Separation of payload
11) Touchdown of payload on surface

COSTING

Masses were obtained from STRC and the percent iheritance classes were obtained from the individual subsystems. The equations outlined in Ref. 9 were used to estimate the cost of the mission.

Total hardware	
four delivery systems	265 million dollars
Total development	430 million dollars
Mission operations	79 million dollars
Total Cost	774 million dollars

TECHNICAL PROBLEM AREAS

Don't know how long aerobrake maneuver will last
Don't fully understand costing equations

NOTES

1 Reference #7, p. 4167
2 Reference #2, p. 67
3 Reference #1, supplimentary handout, orbital elements and physical characteristics of the Sun and planets
4 Reference #6, p. 4297

REFERENCES

1) Prussing, J.E., and Conway, B.A., Orbital Mechanics (Version 2.0 1987). Class notes for AAE 306, University of Illinois.

2) Moeckel, W.E., Trajectories With Constant Tangential Thrust in Central Gravitational Fields. NASA Technical Report R-53, 1960.

3) Jensen, J., et al. Design Guide to Orbital Flight. New York, NY: McGraw-Hill Book Company, Inc., 1962.

4) Breakwell, J.V., and Rauch, H.E., "Optimum Guidance for a Low Thrust Interplanetary Vehicle" AIAA Journal Vol. 4, No. 6 (April 1966), p.p.693-704.

5) Soffen, G.A., "The Viking Project" Journal of Geophysical Research Vol. 82, No. 28 (Sept. 30, 1977), p.p. 3959-3970.

6) Mayo, A.P., et al. "Lander Locations, Mars Physical Ephemeris, and Solar System Parameters: Detrermination From Viking Lander Tracking Data" Journal of Geophysical Research Vol. 82, No. 28 (Sept. 30, 1977), p.p. 4297-4303.

7) Veverka, J., et al. "A Study of Variable Features on Mars During the Viking Primary Mission" Journal of Geophysical Research Vol. 82, No. 28 (Sept. 30, 1977), p.p. 4167-7187.

8) Hollister, W.M., and Prussing, J.E., "Optimum Transfer to Mars Via Venus" Astronautica Acta Vol. 12, No. 2 (1966), p.p. 169-179.

9) Soldner, J.K., "Cost Estimation Techniques for Planetary Missions"

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Viewgraphs from a presentation to AAE 241 Design Class on April 14, 1987, University of Illinois.

10) AAE 241 Class Notes, M. Lembeck, University of Illinois.

11) AAE 404 Class Notes, Proffessor B.A. Conway,University of Illinois.

12) Conversation with M. Lembeck, Section Leader of AAE 241 Spacecraft Design Section, University of Illinois.

13) Conversation with Proffessor J.E. Prussing,University of Illinois.

TOMASO DI PAOLO

STRUCTURES

(STRC)

INTRODUCTION

At the beginning of the 1988 Spring semester, a document for proposal was received entitled "Request For Proposal and Preliminary Design of a Manned Mars Aircraft Space Delivery System." This report listed the following requirements for the Structures Subsystem (STRU) of the delivery system:

(1) the structures subsystem will include the materials, the design , and thermal control
(2) the spacecraft components and payload will be deliveried to orbit in the cargo bay of the Space Shuttle and be assembled on-orbit at the space station spacecraft assembly-and-repair facility
(3) the spacecraft will be retrievable by a remote manipulation device on the space station or the Space Shuttle
(4) Nothing in the spacecraft's design should preclude it from performing several possible missions, carrying vastly different payloads to different destinations
(5) it should not use materials or techniques expected to be available after 1998
(6) the spacecraft will have a design lifetime of four years, but nothing in the design should preclude it from exceeding this limit.
(7) the design will stress simplicity, reliability, minimum mass, and low cost.[1]

Thus, the requirements are to conceptualize a spacecraft that is capable of bringing an aircraft and a satellite to Mars. The spacecraft will be brought to the assembly facility by the shuttle and will be constructed there. The delivery system will be able to carry different payloads, be retrievable, and will provide structural support for the instrument bus and payload entry system.

METHOD OF ATTACK

The structure of the spacecraft is affected by inherited constraints from the RFP and by other constraints received from interactions with the other subsystems. A review of former structures courses provided a familiarization with with structural needs, and then research on space structures was done, with emphasis on accumulating information on monocoque shells, trusses, metals, compositites, thermal protection, and space environment's affects on a spacecraft. Information was also gathered through the lectures on structures. This information resulted in trade studies on thermal control and comparisons of materials, and into the formulation of structural theories in the design of the spacecraft .

INHERITED CONSTRAINTS

There are three dimensional constraints on the payload system PIVOT (Payload Internal Volume Optimized Transport):

1)since the shuttle is to be used, the dimensions of PIVOT cannot exceed the 4.6 m radius by 18.3 m length Orbiter payload envelope[2],

2)if the airplane were to be disassembled then the longest single item of the aircraft would be the shortest length of PIVOT (to be further discussed), and

3)from information gathered by the AERO subsystem, the length and frontal area of a combination of PIVOTS is also constrained (to be further discussed).

PIVOT needs to be simple and light weight but also able to withstand shuttle launch loads of six times Earth's gravity (6Ges) which incorporates a safety factor[3]. This load determines the thickness of the truss members which were used to design a skeleton for PIVOT.

A triangular truss was designed to connect the three PIVOTs, satellite, and engines together to form the spacecraft. The truss was designed to be able to withstand the impulsive launch from Earth to Mars, and hold up to the aerobraking forces to be encountered during aerobraking.

The materials being used need to be protected from the space environment and the satellite and propellant tanks need to be maintained at certain thermal levels regardless of external heat fluctuations.

INTERACTIONAL CONSTRAINTS

The aircraft designed by the aircraft group imposes a dimensional constraint on the payload system PIVOT. The plane was initially broken into parts (4.6m x 13m x 4m) so that the plane could be transported to Mars. Three PIVOT canisters were required to hold the aircraft and so resulted in a triangular stacked configuration of canisters whose frontal area was approximately 24.5 m (Figure 1). After running simulations with AEROB, the AERO subsystem informed me that the length of PIVOT had to be reduced because the wake behind the shield would impact on the canisters (see AERO report). According to AERO, the longest canister can only be 9.3m in length. Returning to the aircraft section, the plane was broken up again and after graphically trying to store pieces into PIVOTs, the aircraft STRCs person and I agreed that the only feasible way to package the aircraft was to disassemble the wing, break the canard into two pieces and also the tail, and transport the fuselage as a whole piece. The longest length of the aircraft was now a 7.5 m rib, so PIVOT has a payload envelope of 7.5m by 4.5 m diameter.

PROP determined the mass of fuel and fuel tanks needed so I designed tanks for the fuel volume and placements constraints. The truss members were designed as PIVOTs skeleton at 6Ges and so the same trusses were used for the triangular truss which has to support the thrust imparted upon it from the PROP engines. When a thrust figure was received by PROP, the maximum load capability of the truss was compared, so that PROP could adjust his thrust and burn time if the thrust loads were too high.

SCIN delivered science instruments with specific tasks to perform (see SCIN report) so the masses, dimensions, and tasks of the instrments determined the satellite configuration.

AACS required the use of accelerometers, gyros, thrusters, and thruster fuel on both the payload and satellite. Some of his other instruments required axial placement and unobstructed viewing. AACS received from STRC the inertia matrix of the satellite in order to determine the principal axes of the satellite.

CCC designed a UHF antenna, UHF transmitter, and communications antenna to be placed and positioned of the satellite. Commputers and tape recorders were also designed to be placed inside the satellite and payload. The antenna placement required as much unobstructed viewing of Earth as feasible during the Earth-to-Mars leg of the mission.

AERO designed the aerobrake with the knowledge of PIVOT frontal area and in return constrained the length of PIVOT. AERO the required STRC to supply a payload lander mass in order to calculate parachute sizes and mass (see AERO).

MMPS interacted with me when I was group leader and spacecraft-aircraft liason, but only required structural masses from me for cost evaluation. Indirectly, MMPS did constrain me due to his interactions with other subsystems by determining trajectories which affected propulsion needs which in turn affected tank masses and sizes needed to be placed on the spacecraft.

THERMAL CONSIDERATIONS AND THE SPACE ENVIRONMENT

During the cruise phase of the mission, the thermal system of the satellite needs to keep the instruments above 68°C (20°F)[4] and the propellant thermal system needs to keep some of the propellant at approximately 310K. The thermal enviroment of the satellite can be maintained by the use of the RTG to power a heater in the satellite and a temperature control device to regulate the internal temperature. the propellant thermal system would use insulaton such as foam, multilayers, fibrous, or superinsulators. Superinsulators are light and have a conductivity of 1.5×10^{-3} BTU/hr-ft^2 -F (at Martian surface pressures), and at a thickness of one inch versus six inches of fiberglass insulation has an equivalent conductivity[4]. Superinsulators are very expensive so combinations of the two can be used to keep cost and weight down. Insufficient research information on the conductivity of insulators in space and insulation densities prohibited a trade study on combinations of insulators vs. coldest environment scenario and calculations of insulation mass. I was able to find a graphical relationship of insulation vs. lander area (in this case PIVOT) which demonstrates how thick of insulation could be required to insulate the tanks and PIVOT, see Figure 2[4]. Another graph demonstrates the use of multilayer insulation (MLI) vs. heat flux at low Earth orbit (LEO) to insulate hydrogen-oxygen propellant tanks, see Figure 3[5].

The number of micro-meteoroid penetraions was an early worry in the design of PIVOT's aluminum skin thickness.. Research information revealed that there was a probability of no penetrations of .99 during the Earth-to-Mars journey, of .995 during the Mars braking stage, and of 1.0 if an areobrake shield were used to brake. The chances of a micro-meteoroid impact are small, so an increase in the skin thickness due to meteoroid impact was not taken into account if the thickness seemed thick enough (engineering intuition).

STUCTURAL DESIGN

Propellant Tanks

The design of the propellant tanks was first determined by their orientation in the spacecraft . The retrorocket fuel was placed in the triangular truss in order to optomize the use of empty space between the cylinders. With truss lengths of 1.1 m forming the equilateral triangle sides of the truss, a .3 radius sphere could be placed within the truss.

RETRO FUEL (ASSUMED TO BE ALL HYDRAZINE, =984 KG/M)

MASS = 340 KG VOLUME = .346M= VOL

RADIUS OF SPHERE=r_s =(3VOL / 4π) = .436m
CYLINDER LENGTH OF 2r PLUS TWO HEMISPHERES (diagram to the right)
r_{2r} = (3VOL / 4π) = .32m

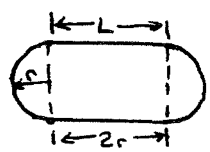

Diagram of an r_{2r} cylinder.

A cylinderical tank of radius .32m by 1.28m long was chosen to be placed within the truss toward the aerobrake. This tank was chosen because it fit and only one tank was needed to be constucted. The fuel for attitude and control during the Earth-to-Mars part of the mission was also placed in the opening by the cylinders in order to use up empty space and to maintain the effective radius of 4.95m.

AACS FUEL (HYDRAZINE): MASS=25.3 KG VOLUME= .0257M^3

r_s=.18m r_{2s}=.13m

A spherical tank was chosen because a sphere will use less material in its manufacturing.

The fuel tank for the satellite was approximated to use 76kg of hydrazine, and has a radius of .26m.

The masses for the fuels used in Mars orbit insertion and the inital impulsive thrust from Earth were too large to be placed in the truss (the length of the cylinders caused the spacecraft to become very long), so spherical tanks were chosen since they would use the smallest mass of material.

INSERTION

Mono. Hydrazine : 662.2kg : r_s=.543m
Oxidizer (nitogen-textroxide) : 1092kg : r_s=.575m

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EARTH-TO-MARS

Oxygen : 1909.8kg : r_s=1.55m
Hydrogen: 6493 kg : r_s=2.79m

The thickness of the spheres can be calculated by solving the stress equation for a sphere or a cylinder [7].

$$\sigma_s = pR_s/2t_s \qquad \sigma_{c_{circum}} = \sigma_{c_{max}} = pR_c/t_c$$

where σ_S and $\sigma_{C_{max}}$ can be set at the yield strength of some material, the radius of the sphere(R_S) or the cylinder (R_C) found above, and the fuel operating at some internal pressure. No information was known concerning operating pressures, so the drawing of the tanks assumed that the tank thickness was small compared the radius.

MONOCOQUE AND TRUSSES

As different parts of the mission were examined, loading due to the shuttle launch, to the impulsive Earth-to Mars launch, and to the aerobraking at Mars were considered to necessitate further examination. The launch at Earth imposed the highest acceleration on any part of the spacecraft, so the most massive single piece of the spacecraft would endure its highest loading here. The largest single piece of the spacecraft is PIVOT so its structure was built to endure an acceleration of 6 Ges (this has a safety factor of 1Ge)[3]. $F=ma=6(9.8\ m/s^2)m$

The next two loads affect the truss "backbone" of the spacecraft. The impulsive Earth-to-Mars phase of the misson uses a single engine racket that produces a thrust of 100,000 N (see PROP) which was designed after the mass of the Mars insertion vehicle was determined. Depending upon the acceleration of the rocket (see PROP), the loads being handled by the truss structure at 1 Ge is ...

$$F=m_t a=a(m_p+m_s) \qquad 100{,}000N=9.8m/s^2(8402kg+m_s)$$
$$m_s=2162 \qquad \therefore Load=2162kg(9.3m/s^2)=21{,}188$$

During aerobraking, the load is due to the drag on the aerobrake.

$$D=.5\rho V^2 C_D S$$

where ρ= the density of the Martian atmosphere is dependent upon the height of travel through the Martian atmosphere, V is the velocity at perigee, C_D is the coefficient of drag on the shield, and S is the sphere's area. The largest drag will occur during the periapse radius of the spacecraft's elliptical orbit, so V needs to be found:

$$V=(u(2/r_p-1/a))^{1/2}$$

where r_p=periapse radius, a is the semimajor axis of the spacecrafts, and u for Mars is $4.305\times10^4 km^3/s^2$. By solving for the velocity at periapse, the drag can be found.The load is not as high as the impulsive launch, so the impulsive load will be the constraint on the truss.

PIVOT was designed to have a truss skeleton which would carry the major load of shuttle launch. The truss members or tubes were constrained to a 5cm distance (the difference between the shuttle payload radius and the radius allotted to the aircraft package) and a length of 7.6m

(a fixed length of PIVOT). The tubes each have a length to radius ratio, or slender ratio, which will result in failure due to buckling before failure due to "shell buckling or compressive/tensile ultimate strength failures."[8]

The same failure is true for the truss members holding the containers together. "Using the Euler buckling formula which for pinned end columns is defined as $P_{cr}=\pi^2 EI/L^2$ where P_{cr} is the buckling load, L is the length, I is the minimum moment of inertia,...,and E is the longitudinal modulus of the tube,"the thickness of the truss members can be obtained.[8]

A monocoque shell was used for the design of PIVOT's outer shell since the loads would be primarily carried by the truss skeleton. The equation for the axial stress in an unpressurized thin wall cylinder is given by $\sigma_c/E=9(t/R)^{1.6}+.16(t/L)^{1.3}$, where σ_c is the buckling stress, E is the modulus of elasticity, t is the thickness, and L is the length. This was also used for the satellite wall thickness but was rejected in favor of the tube wall thickness estimate.

MATERIALS TRADE

Material	E $(lb/in^2)\times10^6$	$\rho(lb/in^3)$
Al 2219	10.5	0.10
Al-Li	11.3	0.093
Ti-6Al-4V	16.0	0.1595
Graphite Epoxy	21.0	0.056
P75/ERLX1962	75	0.065
P75/ERLX1962-Al Clad	40	0.074
6061 Al	9.9	0.098

MATERIALS USAGE

Using the unpressurized thin wall cylinder approximation at a load of 6 Ges, the best material to use for the PIVOT wall was 6061 Al which resulted in a thickness of 1.09×10^{-4}m and mass of 4.27 kg.

The thickness for the longest member on PIVOT determined the thickness of all the struss members since this member is under a load of 6 Ges and will buckle before any of the other truss members. PIVOT's skeleton has short members (.3m) forming the ring about the cylinder and

1.906 m diagonal trusses went diagonally down the side to carry the axial load. Using the Euler buckling formula and a loading of 6 Ges, EI was determined to be 5.21×10^6 lbin^2. Looking at figure 4a [8], the thickness of 0.04in. (1.016×10^{-3} m) was chosen for the material P75Gr/Ep. Now using the formula Mass/Length=$p(r_o^2-r_i^2)$ (see also figure 4b), the mass per unit length was found to be .297 kg/m. Adding up the 172 truss members resulted in a total mass of 120.3kg. Figure 5 shows PIVOT and the nodal location of the trunions which are connected at the nodes of the trusses to carry the loads axially to the truss skeleton The trunions are able to be retrived by a remote manipulation device duto their size. The aluminum skin on PIVOT is primarily to keep radiation from all the graphite truss members, to protect the aircraft from heat fluctuations, and provide some micro-meteoroid protection.

The "backbone" of the spacecraft is the triangular truss which connects the aerobrake to the engines with PIVOT and the satellite also connected (see figure 6). The entire length of the truss is 14m with 24 equilateral truss side members of 1.1m in length, 21 longitudinal members of 2m length,and 42 diagonal truss members of length 2.06m. The diagonal trusses connect and carry the mass of the PIVOTs during the impulsive launch. Since the diagonals are the longest members and connect to PIVOT, they will be assumed to buckle first. Thus, a worst case scenario of only three dliagonal members(only one member carrying a load at each mode of a section) was assumed in order to strengthen the struss without relying on a yield safety factor of 1.4. The axial loading during the impulsive launch results in 100,000 N (1.4)/3=4.67×10^4N being carried by each member, and results in an EI of 2.00

65×10^4 Nm^2 (6.99×10^6lb in2).
Referring back to figure 4, a thickness of 0.05in. to 0.17 in, could be used depending upon the Young's Modulous,E, of the material. A good material for the truss is P75 Gr/Ep- Al clad which has a 4mil of aluminum coating on its surface in order to protect the carbon from space radiation, the thermal environment, and oxygen contamination while in LEO assembly. The total length is 154.92 m and the total mass is 48.03kg.

The six lander legs for PIVOT were built using the same material as for the truss, and have a total mass of 2.8kg. The supporting rods between the cylinders use the same material but have a length of 5m. Using the maximum load for the space station of 5337.6N (1200lb), the EI for the supports is 1.35×10^4Nm^2 (4.7×10^6lbin^2) which figure 4a shows as being in the load range for the designed member.

SATELLITE

Using the monocoque theory for the satellite resulted in a thickness of $t=4.5118 \times 10^{-5}$m and a total mass of approximately 4.3 kg. This is a poor structural design eventhough the walls will hold up to the loads of 6Ges, and twisting of the wall will cause it to fail immediately. The thickness of the truss members was adopted to the wall thickness and resulted in a total weight (outer wall+covers+inner wall) of approximately 100 kg. This is heavy but will easily handle launch conditions and will not crumble under a lateral load.

The reason for the satellite is to make scientific observations of Mars and to help the aircraft during a rescue mission. The placement of the science instruments to support their viewing requirements governed a major portion of which instruments went where. The "backbone" truss runs right thru the middle of the satellite and elongates the width of the satellite (see figure 7). This turned out to be a plus and not just an oddity. A majority of the science instruments needed to be pointed at Mars (see SCIN for specific instrument pointing needs) with the radar altimeter and pressure mirror pointing at the surface while the scan platform on a gimbaled arm allowed the infrared spectrometer, IR Mapper, and UIUC to point at specific locations . Two instruments needed their own personal booms which dictated the length fo the RTG boom. When the satellite is free of the rest of the spacecraft, the truss through the center will remain since it is a strong sructural spot to attach the booms and provides a place for the AACS fuel tank. The antenna was initially boomed but was placed on a tripod support structure after an initial running of INERT showed that its placement on the tripod would help raise Ixx and lower Iyy of the satellite (see AACS), increase the viewing capability of the antenna capability during the Earth-to-Mars leg of the mission (see CCC),and reduce the structural worry about antenna boom vibrations and strengthening of the boom, and /or tha possibility of the boom twisting off due to the extra mass of the USA transmitter on the antenna focal point.

AACS instruments were positioned above the SCIN instruments which resulted in a good final center of mass and met the pointing requirements of the instruments (see figure 7 for the inertia matrix).

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PROBLEMS

-The inertia matrix vlues about the Ixx and Izz are very close and could result if the satellite flipping from one axes to another .

-The backbone truss needs to be analyzed further in order to determine structurally weak members and the principal load carrying members.

-A problem appears concerning the attachment of the inital engine and fuel tanks to the main truss.

FINAL DRAWING

Figure 7 is the drawing of the initial configuration of ARES.

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1)RFP Distillation,AAE 241,1988.

2) STS User Handbook, NASA, May 1982.

3) Joe Santos Lecture, March 22, 1988.

4) Rosenberg, Morton Jacob, "Thermal Considerations of a Landed Vehicle on the Surface of Mars," Progress in Astronautics and Aeronautics, Vol 23, 1970, pp 49-51

5) Brown Norman, Mars Transit Vehicle Thermal Protection System: Issues, Options, and Trades, NASA-MFSC, 1988.

6) Fuller, Stan, Mars Landing Mission: A Structural Approach, Marshall Space Flight Center, 1988.

7)

8)

Fig. 1

This is a diagram of the area of PIVOT and how they combine to form an effective circle.

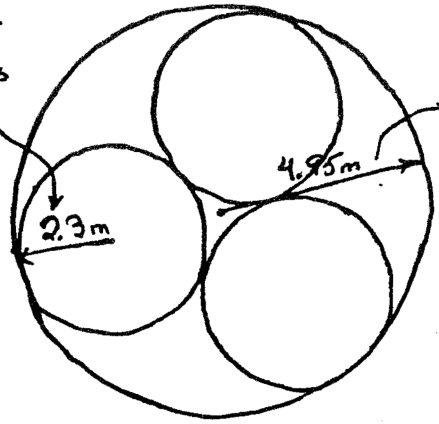

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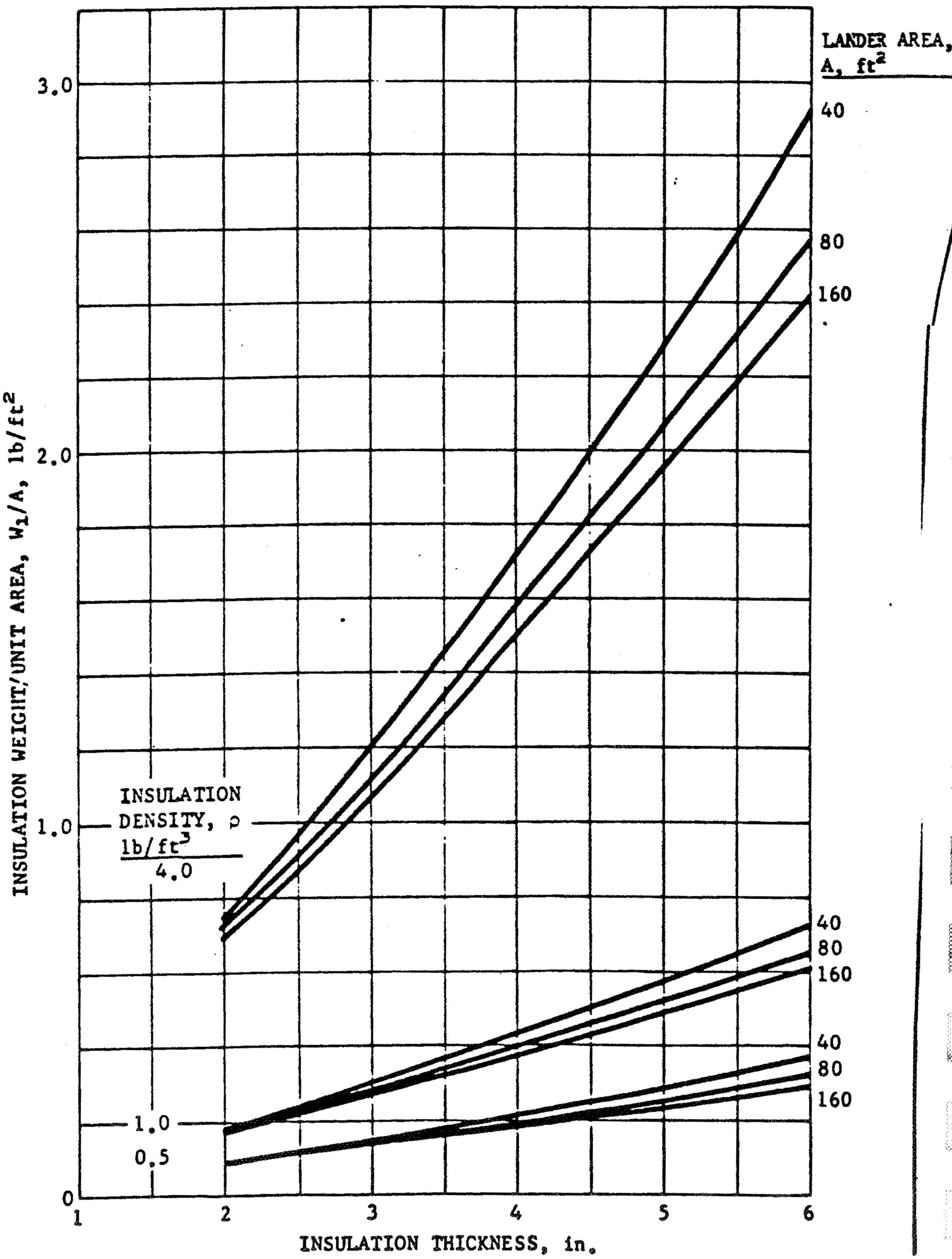
Fig. 2

Insulation Weight vs. Thickness



64-ARES PROJECT
STRC

CHART 2:1

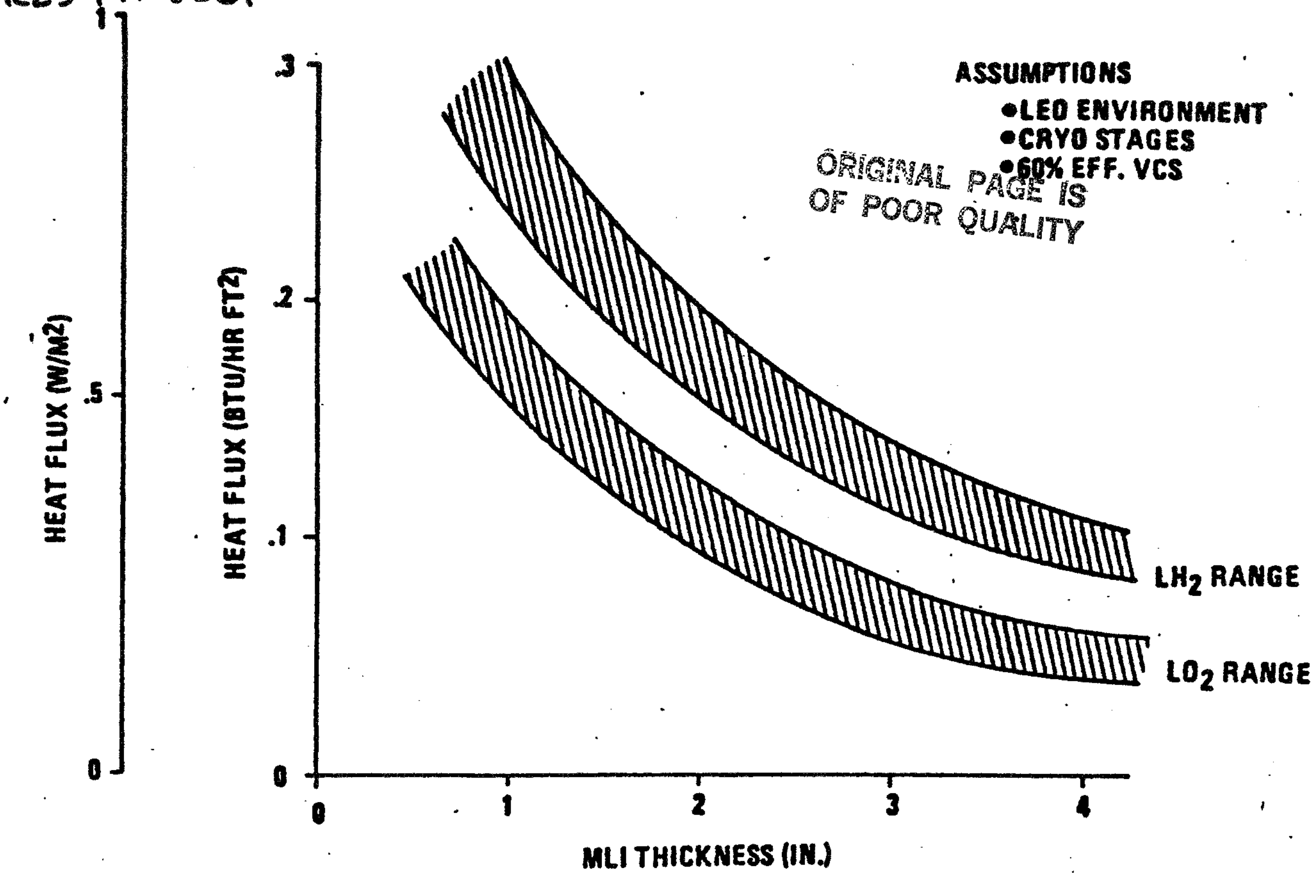


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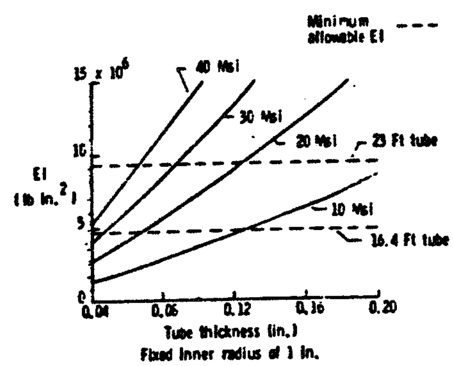
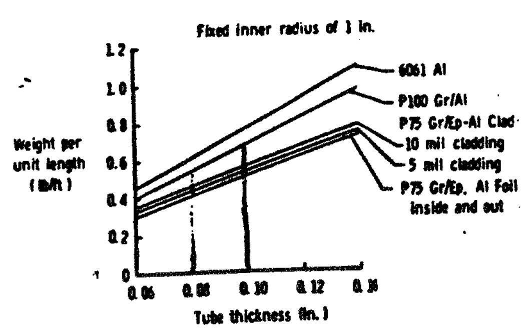
Figure 3: **MARS TRANSIT VEHICLE HEAT FLUX VS. MLI THICKNESS**

Figure 4a:



Flexural stiffness (EI) as a function of tube wall thickness.

Figure 4b:



Weight per unit length (W/L) of various materials as a function of tube wall thickness.

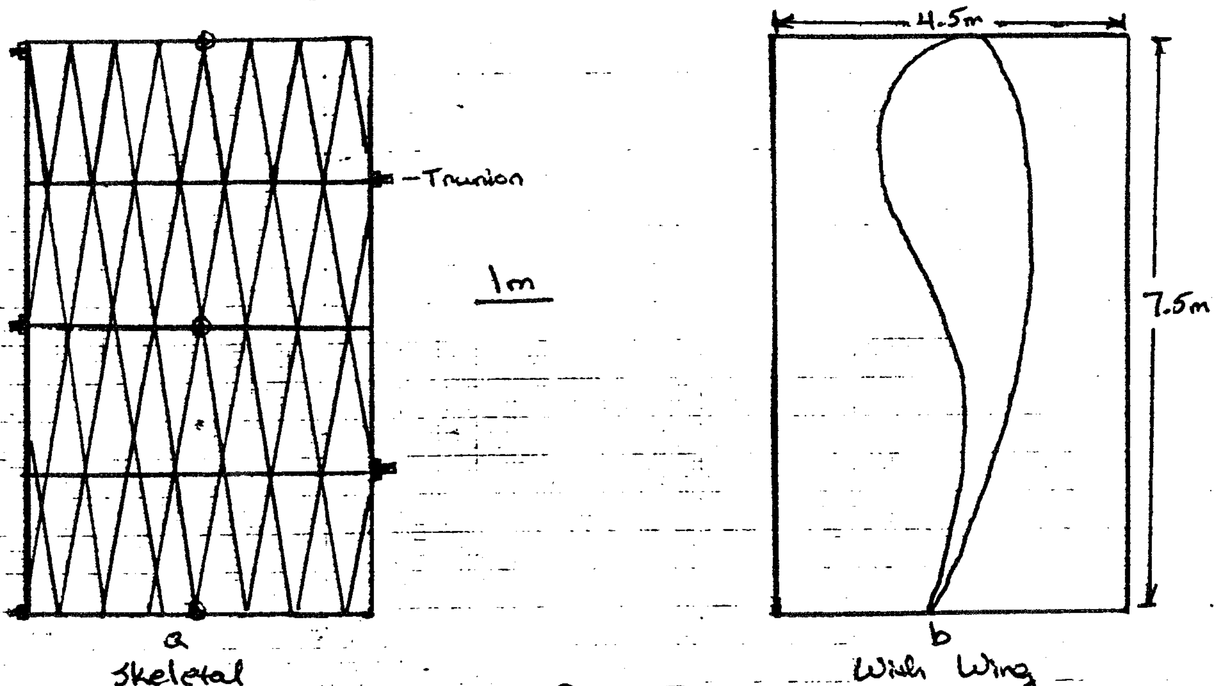


Fig. 5 PIVOT

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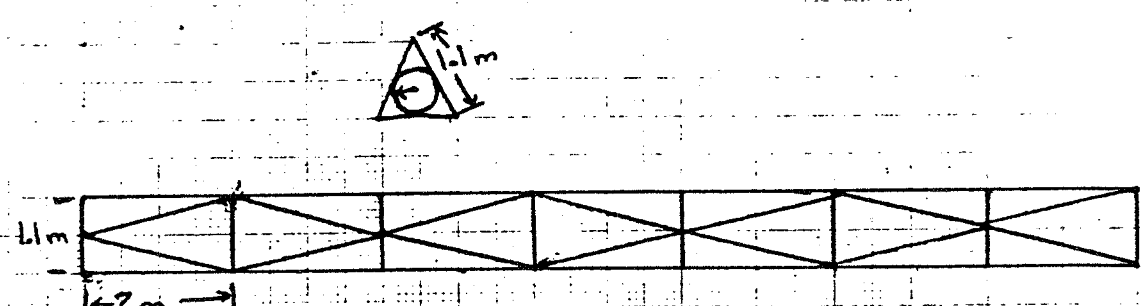

FIG. 6 MAIN TRUSS

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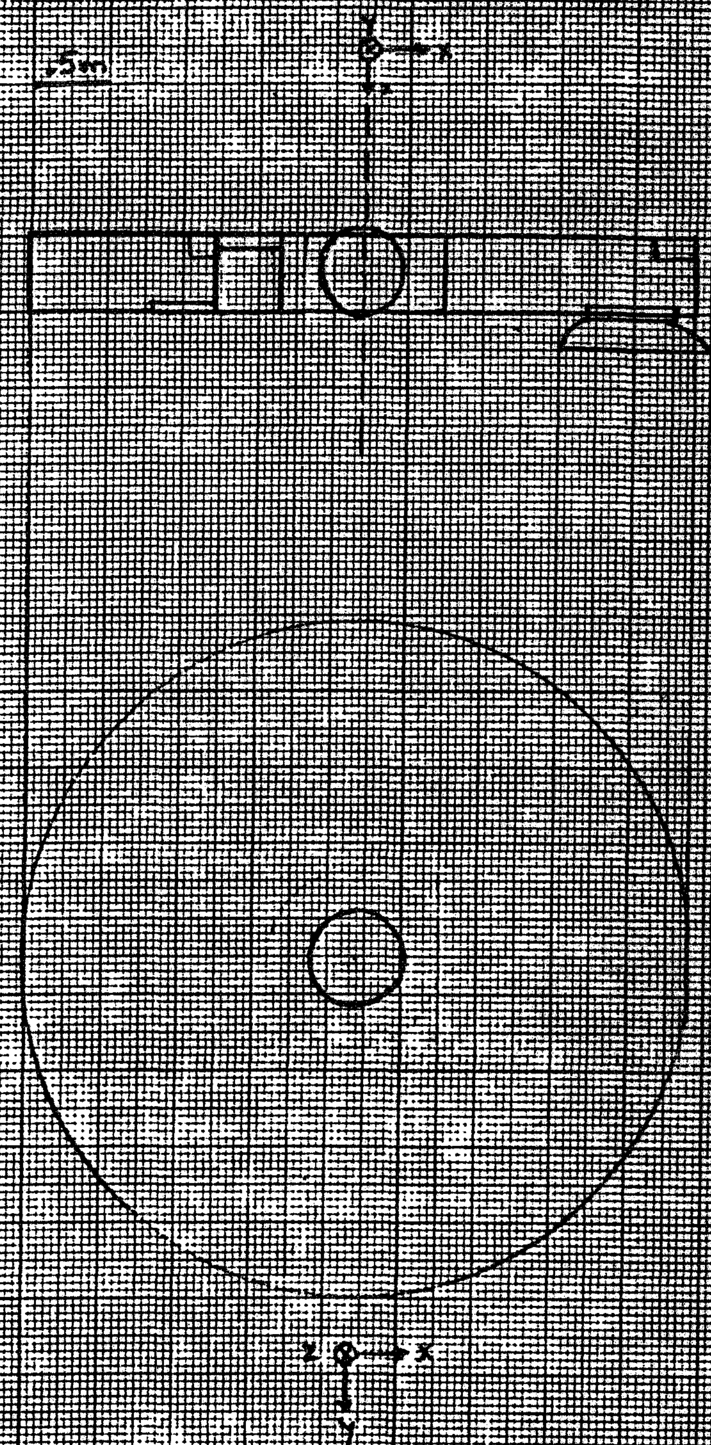

Fig. 7 Inertial Placement of components on a satellite.

$$\begin{bmatrix} 817.6 & -0.1 & 1.7 \\ -0.1 & 642.4 & -115.7 \\ 1.7 & -115.7 & 784.5 \end{bmatrix}$$

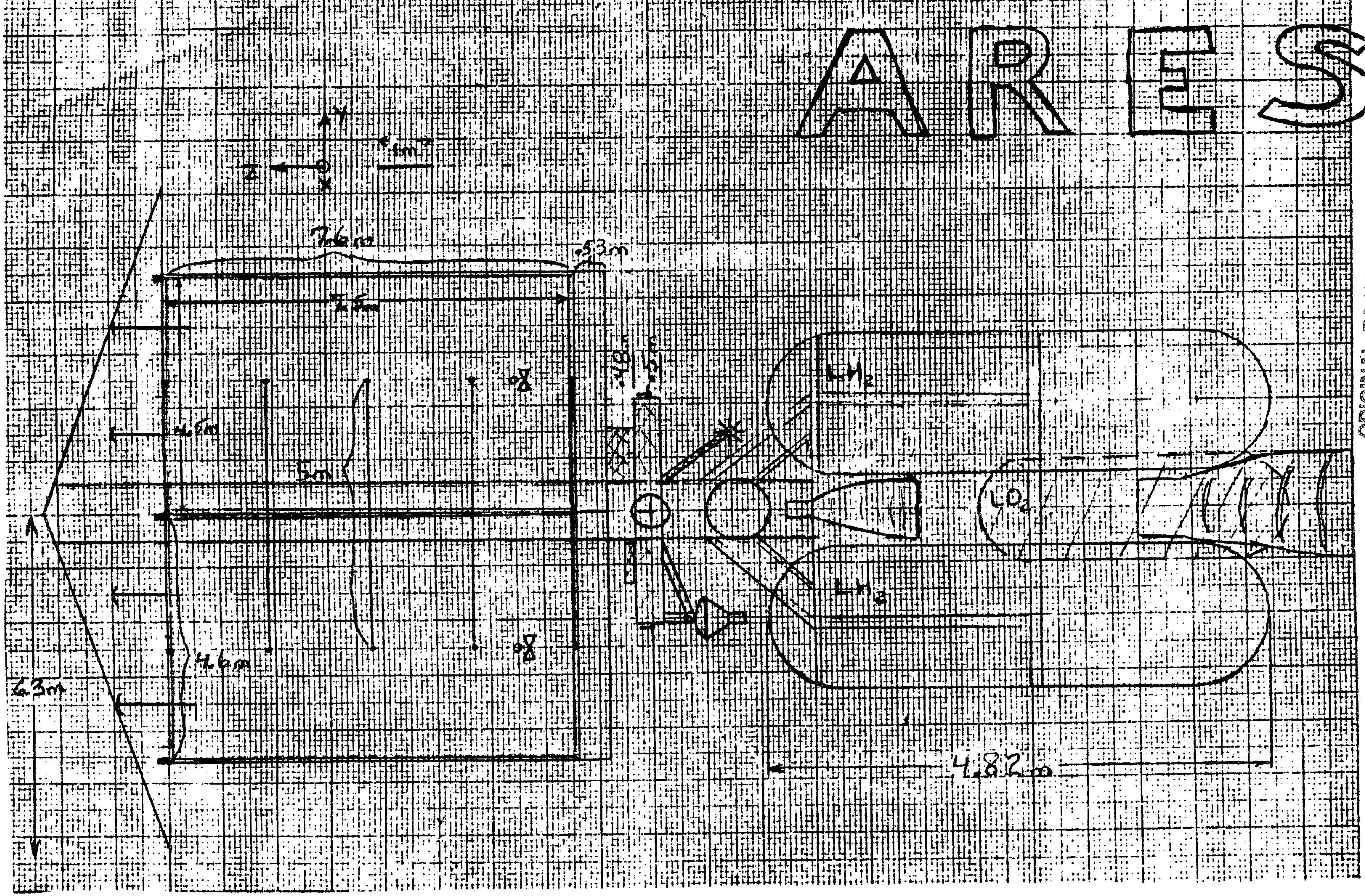

Figure 7: Spacecraft Configuration

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I. Introduction: Power and Propulsion Subsystem

In suppport of the ARES mission, the power and propulsion subsystem was required to satisfy or operate within the constraints of the Global Mission Requirements outlined by MMPC (Mission Management, Planning, and Costing), as well as satisfying the following Power and Propulsion Subsystem requirements:

1) Control Power Relays.
2) Control Valve Actuations.
3) Support Aerobrake Subsystem.
4) Support Structure Subsystem.
5) Support AACS - Maneuvering Control and Power.
6) Support Command and Data Control Subsystem - Power.
7) Support Science and Radio Relay Subsystem - Power.

The trade objective for this mission is to **Minimize** system **Mass** and **Complexity** to **Minimize Cost** and **Maximize Reliability**. To better facilitate the representation of the design, all information will be given as it was traded and designed. The mission was designed in reverse sequence from the lander back to initial Earth launch.

II. Payload/Lander

A. Propulsion.

The lander will use the Aerobrake for thermal protection, chemical thrusters for control and a chemical rocket for powered decent to the surface. Since an unpowered landing is not possible due to surface topography and payload sensitivity, the remaining option is a powered, "soft" landing on Mars. The parachutes detach at 1500 meters altitude with a lander velocity of 45 m/s (See Aerobrake). A soft landing assuming a constant mass and constant thrust resulted in a 67 second burn time, with a total thrust of 13,630 N (3098 lbf) -- See Enclosure (1) for detailed derivation. Four (4) engines, throttable from 12% to 100% power, permit significant attitude and control (See AACS) and are well within current technology and compatable with the Space Shuttle RCS thrusters[1]. A Fuel Mass safety factor of 3% was assumed as well as a tank and plumbing mass factor of 10%. Frictional drag and gravity was assumed to provide a positive feed of hydrazine to the decent engines, thus eliminating the need for a blowdown or pump feed system[2]. The following propulsion data does not include AACS.

LANDER PROPULSION DATA

Engine mass	11.5 kg
Mass flow rate	1.155 kg/s
Specific Impulse	301 s
Fuel Type	Hydrazine (N_2H_4)
Tank mass	31.4 kg
Structure mass	15 kg
Fuel mass	314.5 kg
Total mass	406.9 kg

B) AACS

All attitude and control design was done by the AACS subsystem. Integration between AACS and PPS only covered fuel system compatability and feasabilty. It was determined that control of the lander would be accomplished using the decent engines as discussed above as well as twelve (12), ten (10) N thrusters for roll and lateral control. For specific design considerations and thruster placement , SEE AACS p. The final data is presented here for reference:

LANDER AACS DATA	
Fuel Type	Hydrazine (N_2H_4)
Fuel Mass	25.3 kg
Tank Mass	5 kg
Support Mass	10 kg
Thruster Mass Total	18 kg
Total System Mass	58.3 kg

C) POWER

The lander had power requirements totaling 32.8 We. This was derived from the Science Subsystem-- (See SCIN) and Command, Control, and Communication Subsystem-- (See CCC) requirements. Power for valve actuations was considered minimal, since no power data for such systems were mentioned in the literature examined. An additional 1% power factor was assumed with an operational lifetime of 57 W-hours upon lander separation from the bus, allowing adequate data collection and location transponder transmission. Advanced Ag-Zn batteries were used based on a high power to mass ratio of 110 W-Hr/kg, versus a power to mass ratio of 22 to 26 W-Hr/kg for higher life cycle batteries (Ni-Cd)[3]. Cycle life is not relavent to this once discharge requirement, therefore this state of the art battery can be used. The total battery weight is 2 kg including structural support and wiring. Assuming the density of Zn as a rough battery density with extra room for error, the battery volume is 150 cm^3. No attempt was made to design an elaborate electrical support system for such a detailed design area, however, a listing is presented: (See SCIN and CCC for details).

LANDER POWER DATA	
Retarding Potential Analyzer	2.8 W
Mass Spectrometer	13 W
Auto Altitude Eval	5 W
Space Transport	5 W
Tape Recorders	1 W
Computer	1 W
Antenna	5W
UHF	5W
Peak power	32.8

III. Instrument Bus

A. Propulsion.

After initial lander design, the next phase to consider is Mars Orbit Insertion (MOI). The following constraints have been placed on the method of MOI by various Subsystems and the RFP. The Subsystem will be listed following the constraint:

1) Aerocapture rejected due to heating and control considerations (AB).
2) Aerobrake required by RFP, therefor Ion Engine not reasonable option for MOI. (RFP)
3) Three Axis Control has been chosen at Mars (AACS).
4) Thruster controls for manuevers are required. (AACS).
5) Burn duration should be minimised for circularization burn (AB).

These constraints lead to the necessary selection of either a solid or liquid chemical propulsive burn. The advantages and disadvantages were traded to determine type of system[4]

.

TRADE:

SYSTEM	ADVANTAGES	DISADVANTAGES
SOLID PROPELLENT	Impulsive Maneuver Well Tested Easy to store and handle	Lower Isp (290) One shot ignition Cost effective for spin stabilized
BI-PROPELLENT**** (Nitrogen tetroxide-monomethal hydrazine)	High Isp (310 s) Cost effective 3 axis Restart Capability Dual use or fuel State of the art	Not fully impulsive unless large engine size Fuel Slosh Propellent mixing
MONO-PROPELLENT (Hydrazine)	Dual use of fuel Restart Capability No fuel mixing	Lower Isp (220 s) Fuel Slosh Not fully impulsive

**** SLECTED DESIGN

The Bi-Propellent Design was chosen because of the inherent advantages of MOI flexibility, the optimization with 3 axis control, and high Isp. The problems of Fuel Slosh and propellent mixing have been delt with on previous missions[5] and do not represent significant problem areas. The trade did not require detailed numerical analysis since the peculiar system characteristics were more valuable than any small mass differences. This satisfies the RFP requirements stressing reliability and also reduces the bus mass, over other methods, through the higher Isp.

3-4 The information for the tankage factors, Isp, engine mass, etc. was given to AB to perform the aerobraking manuever, which was to produce the desired result and provide fuel and mass numbers for subsequent optimization. Problems were discovered with the AEROB program having rather strange data in the data files. Since the Aerobrake had a larger than optimal size needed to cover the payload, the weight of the shield was determined by AB independent of the program -- (See AB). MMPC computed the required delta V's the resulting fuel mass and engine data was calculated --(See MMPC). For a detailed derivation of the engine, fuel, and structure data --- See Enclosure (2). Spacing and sizing of tank are shown in STRUC and the final results show:

MOI VEHICLE DATA

Final orbit mass w/o fuel and tanks	4494.0 kg (90 kg engine)
Fuel mass for MOI (total)	1754.8 kg
Tank mass and structure	175.48 kg
Initial MOI mass	6424.6 kg

MOI ENGINE DATA[6]

Thrust	26,700 N
Isp	310
Mixture Ratio	1.65 (m ox/m pro)
Fuel	MMH (monomythel Hydrazine CH_3NHNH_2
	NTO (Nitrogen Tetroxide N_2O_4
Mass	90 kg

MOI ENGINE BURNS

Apogee from Hyperbolic	1352.0 kg fuel	153.9 sec
Initiation of AB sequence	6.6 kg fuel	0.7 sec
Final Circularization	395.9 kg fuel	45.1 sec

B) AACS

The AACS subsystem was responsible for attaining the proper specifications on control thrusters and fuel requirements, however fuel selection was shared. The detailed description of the system is described in the AACS section, however pertinent data is presented here for reference.

AACS MOI BUS DATA

Number of thrusters	16
Thrust	1 N
Isp	220 sec
Mass flow rate	0.464 g/sec
Mass of Fuel and Tanks	91 kg
Structural Mass	15 kg

Since the aerobraking maneuver was required, the main constraint on our power selection as the aerobraking sequence. The two possibilities were solar and Radioisotope Thermoelectric Generator (RTG). It was also assumed that the bus power system would be used for the Mars Orbit Transfer (MOT) phase if chemical was used, but would only power the bus if ion or some other "high power" MOT system was used. It was determined (with help from MMPC, AACS, and STRUC, and AB) that the aerobraking sequence would require retractable solar arrays and retraction mechanisms of high reliability, battery storage, frequent pointing, etc. Since the power requirement from SCIN (p.XX) was only 125 W peak power, RTG's were researched[7]. In an effort to maximize reliability and reduce complexity (SEE RFP REQ), an RTG was chosen as the bus power source. This eliminates the problems associated with aerobraking, solar array orientation during mission, and eclipse energy storage batteries. The AACS system also saves a large fuel load by not requiring frequent solar pointing maneuvers. For the bus power requirement of 125 Watts, a General Purpose Heat Source (GPHS) RTG fueled with $^{238}PuO_2$ will be used. The GPHS-RTG has a 28 kg mass and is shaped like a cylinder with a 18 inch diameter and a length of 26 inches. For a detailed analysis of the Solar Array, RTG Trade--See Enclosure (3)[8].

IV. Low Earth Orbit to MOI.

INITIAL TRADE[9]

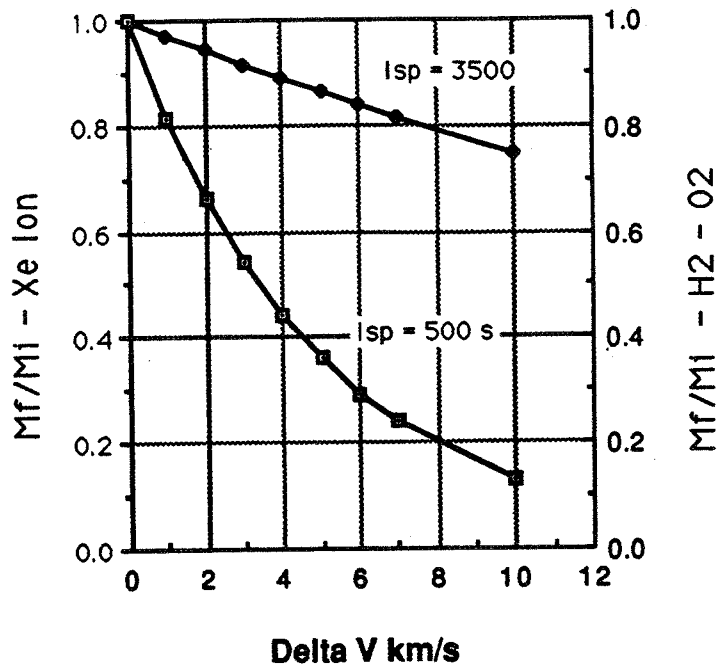
SYSTEM	ADVANTAGES	DISADVANTAGES
CHEMICAL H_2-O_2 **** Isp = 500 s	Previously use/ Reliable Allows Ballistic Trajectory Shortest Duration Missions	Low Isp compared to non chemical Explosive - Launch hazard Storage difficulties
Xe ION**** Isp = 3500 s	Inert Fuel (Xe) High Isp Current Working Model	Large Power Requirements Low thrust/ Mass flow Non Ballistic Trajectories
RESISTO JET Isp = 290 to 380 s	Mid range Isp	In Research Phase
ARCJET Isp = 400 to 1500 s	High Isp	In Research Phase
MAGNETO PLASMA DYNAMIC (MPD) Isp = 2000 to 8000 s	High Isp	In Research Phase

SELECTED DESIGN ****

FINAL TRADE

Since both options presented potential as as excellent propulsion source, an indepth trade was conducted to determine the best choice. The different possible ion combinations were compared to the chemical option. Also, since Ion is strongly coupled to Power the trade includes the Power Trade as well. The following graphs result:

Figure 1 shows the advantages of the high Isp associated with Ion engines. For our delta v of 3.5 km/s the Mass Final/ Mass Initial is 0.9 for Ion and 0.45 for the Chemical case.



Although very encouraging, the above results do not show the hidden propulsion mass dur to electrical/solar power requirements. Since the engines require 5 kW to operate, most of the propulsion mass is in the solar arrays. For this design, solar arrays were chosen over other possible options due to low mass and non-nuclear components. It is the only logical choice until there is a nuclear device caparable in output (kW/kg) with solar arrays. (Note that this mission requires a Mass Final which is still too small to be considered for a space nuclear reactor even if one was available, which it is not.) Ion engine data and solar array data was collected and compared to data used to do similiar trade studies. The equations in Ref (10) were modified to represent current technology possibilities-- See Enclosure (4) -- to get the following results:

Mass Propulsion (dry) (kg) = (34 + 0.34Mp + 250) N + 10

where-- Mp= Mass of Propellent

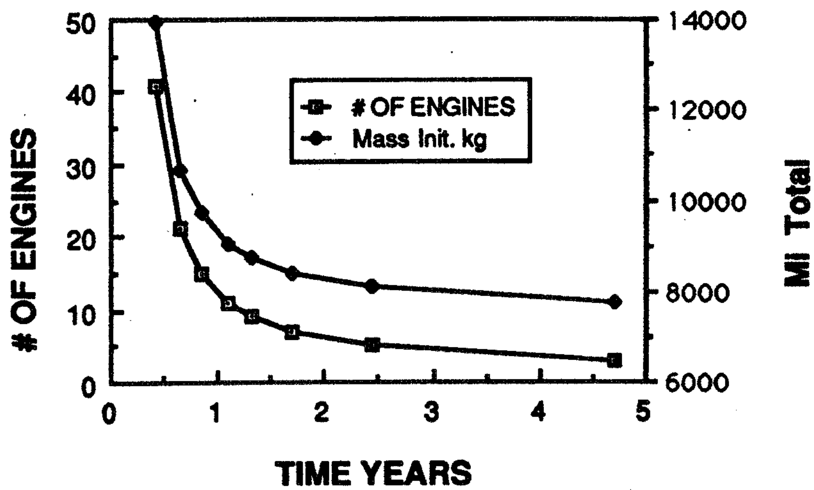
N = Number of Engines/2

A low thrust equation was developed by MMPC and used to calculate the time from Low Earth Orbit (LEO) to Earth Escape.

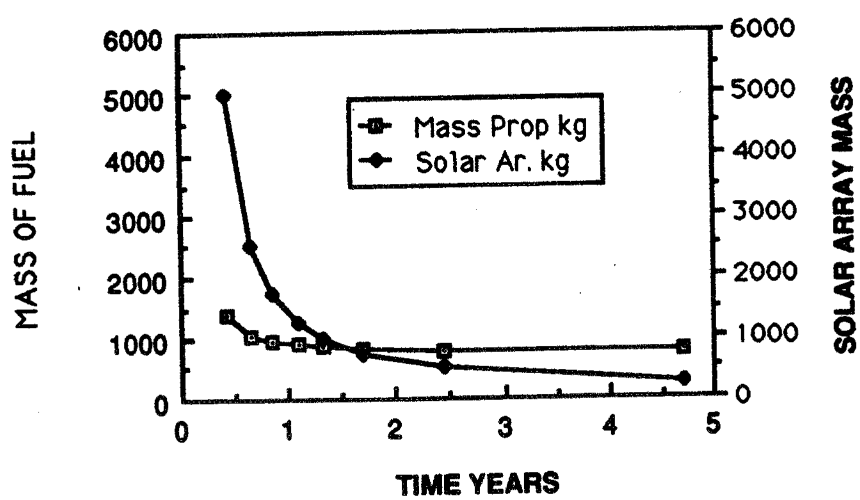
Time (days) = 0.221 (Mass Initial/N)

The following graphs show comparisons of the different combinations considered and how they affect mission duration and planning:

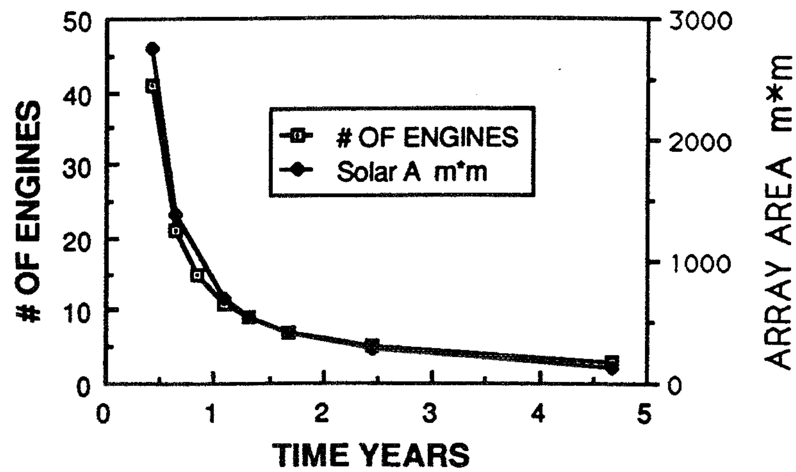
Figure 2 shows a substantial increase in Mass Initial due to increased propulsion requirements to increase thrust and decrease time. Note the dramatic increase in the number of engines as time decreases.



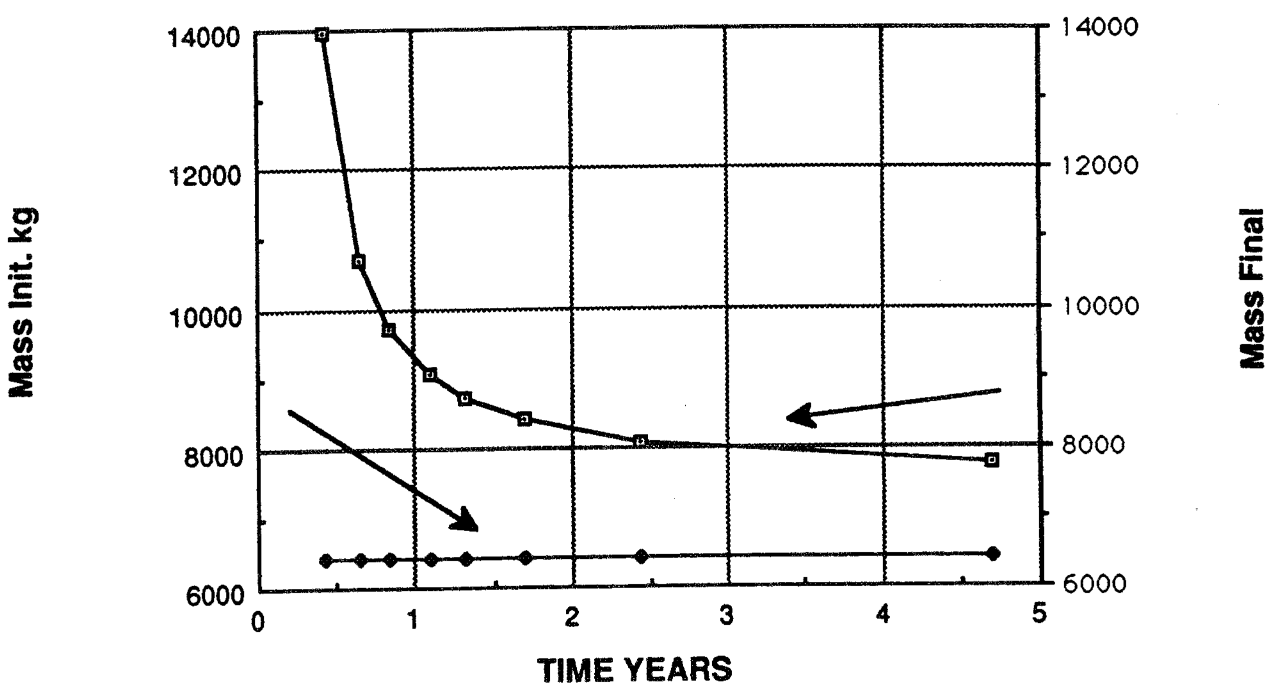
Although propellent mass increases somewhat and engine mass increases very little, the solar array mass does. This emphasizes the dependency of thrust on solar array weight and power. Figure 3.



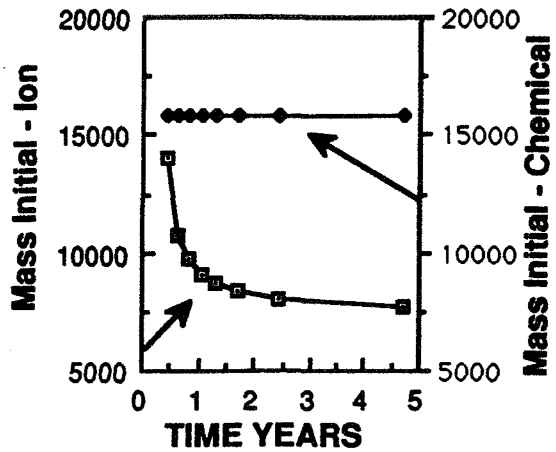
The increase in power follows directly from the increase in engines. Note that the large increase in array area will also impact greatly on the array power to mass ratio. The AACS support will also increase the mass significantly due to the large moment of inertia and increased mass and fuel due to pointing. Figure 4.

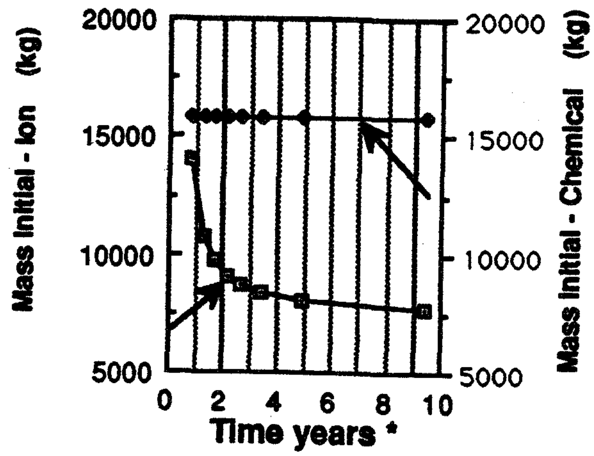


The above results in a final comparison of Mass Final and Mass Initial. It is readily apparent that the requirement of a short duration mission significantly reduces the advantages of Ion propulsion. Figure 5.



When compared to the data obtained from a Chemical LEO to Earth Escape, one still gets a lower initial mass using 40 engines (w/ 1 redundant) and a solar array greater than 2000 square meters (Figure 6). Figure 7 is shown as a comparison to the constant thrust case. Due to solar array eclipe and warm up considerations for the engine the time of mission was doubled.





Even though the mass is less than the mass for the all chemical propulsive case, it was determined that the large number of engines and large solar array size would make the project very complex for AACS, and require high reliability, increased structural problems, low thrust orbit difficulties due to the non- continuous thrust. The small mass savings was not enough to warrent dealing with the complex problems this would present. Matching with the RFP requirements, a simple, well tested and used propulsion system was used, Liquid Hydrogen-Liquid Oxygen. It is important to mention, however, that if the time constraint of the mission was relaxed to the "greater than 4 year" mark, ion should be reconsidered.

Chemical was chosen due to high reliability, short mission time set by the RFP to complete mission within 3 years, impulsive orbit calculation using MULIMP, and simplicity. The specifications for the booster are listed as follows:

LEO TO EARTH ESCAPE BOOSTER DATA [10]

Mass of MOI Vehicle	6424.3 kg
Mass of Propellent	8403.3 kg
Mass of Structure	840.3 kg
Mass of Engine	153.5 kg
Mass of Total System	15821.4 kg

V. Technical problems / Considerations

A. Propulsion.

The initial design study considered all aspects of the design in an overview fashion, but still did not really dig for details. Engine sizes were chosen using linear mass scaling, using one engine mass for the entire relation. The main booster was designed using very rough comparisons with other data from other engines. It could be said that the fuel data is very accurate, while the engine specs are "somewhere in the ballpark". Although this is not necessary for the initial design, it does leave something to be desired when calculating burn times, and interacting with STRUC to determine support loading requirements. Also, more details need to be decerned about using off the shelf hardware for the propulsion mission. Significant cost savings might result from using components from the Mariner and Viking missions rather than new designs.

The design time constraint of this mission needs to be reevaluated, and is the major obstacle to a low mass mission design. It is recommended that future design analysis reconsiders the ion propulsion option. It is certain that this mission has the potential to be radically different if the RFP did not constrain the mission with an Aerobrake system or a short mission duration.

B. AACS.

This area was mostly handled by the AACS subsystem. Since many thrust requirements changed through mission design, it was felt that PPS should delegate the AACS components to the AACS subsystem and act as an advisor rather than the design specialist. More attention would be required later on in the detailed design process.

C. POWER.

The major technical problem is the choice between RTG and solar arrays. Some type of retractable solar arrays need to be investigated to be available if in the future RTG's are unavailable. Exact scaling for the RTG was not possible, and power integrations were not done due to the depth of the study conducted. Also the launch protection system for the RTG during SHUTTLE needs to be considered, as well as Radiation handling at the Space Station. This was assumed to be feasible. Themal requirements for handling the RTG were also sparse and it assumed that no other cooling is required other than radiative heat transfer.

Valve actuation power was also considered minimal since no major sources covered it. This definitely needs to be determined in future analysis to determine impact on overall power system design and/or selection.

Notes

1. Ref. 8, p.44.
2. Ref. 8, p.46
3. Ref. 14, p.349
4. Ref. 8,10,11,13.
5. Ref. 11, p.61.
6. Ref 13, p.195-200., Ref. 8, p.44.
7. Ref. 2,3,4
8. Ref.3,p517.
9. Ref. 1 - 14.
10. Ref. 13.

***NOTE--- All constants and fuel data were taken from Ref 13, except where otherwise noted.

VI. ENCLOSURES

ENCLOSURE 1

Lander mass without tanks, engines, support, etc. = 2231 kg
delta v = 45 m/s (initial velocity when parachutes detach).
Altitude at detach point approximately = 1500 m.

$x(t) = 1500 - 45t + .5(a-g)t^2$
$= 1500 - 45t + .5(T/m - g)t^2$
$v(t) = -45 + (T/m - g)t = 0$ for Vf at x=0

Set x(t) = 0 and solve for t in terms of T,m, and g.
$(T/m - g) = 45/t \quad g = 3.749$

Therefore, $0 = 1500 - 45t + .5(45)t \quad t = 67$ sec
$T = (0.675 + g)m \quad T = 9870$ N

mass flow = T/(Isp g)   Isp = 220   mass flow = 4.58 kg/s   Mass fuel = (m flow*t)= 306.7kg.

From Reference 8, A RCT (SS) R-40A has a thrust of 872 lbs and weighs 13 kg.
therefore total mass of engine = 9870/(3880.4N/13kg)= 33kg.
now add mass of fuel to engine and recalculate. New payload = 2600 + .5fuel = 2453
T= (4.424)(2453) = 10827.5N
mass flow = 5.02 kg/s * 67 sec= 336.5 kg fuel.

Engine size is .1 of size for a typical MOI engine from reference 13. = .1m * .2m.
Engine specs are: 4 engines; mass= (T*.25)(13/3880.4N) = 9 kg.
Thrust = 2706.8 N; Isp = 220s; fuel = hydrazine; mass flow = 1.3kg/s.

ENCLOSURE 2:

From MMPC we get delta v's of 717.89 m/s; 3.937 m/s ; 247.23 m/s
Mf = mo EXP (-delta v/c)   where Isp = 310s,   c = 3038m/s
After each manuever, the new mass and fuel mass is give in terms of mo.

initial mass mo
delta v1 0.2105mo fuel
0.7895mo
delta v2 0.0102mo fuel
0.788519mo
delta v3 0.06162mo fuel
0.7269

mass fuel = 0.27314mo
mass tanks = 0.027314mo
0.3005 mo

Now mo(1-0.3) = Mae+Mpay+Meng

Maero= 1349kg; Mengine = 90kg

Mairplane= 2679kg Mbus = 376 kg

therefor mf=4494; mo= 6424.6
use above data to get delta v's

Assume engine data from ref 13 for Shuttle OME T= 26,700N, Isp = 310s, Mixture rato =1.65 Use trust/kg ratio from referenc 1 = 298.5 N/kg therefor Mass of engine = 90kg and total fuel = 1754.8 kg using a 10% tankage/plumbing factor, Mass of tanks = 175.5kg. T/gIsp=mass flow= 8.785kg/s

Now calculate burn times: Fuelper delta v / mass flow = burn time.

ENCLOSURE 3.

Reference 2 gives a working GPHS RTG output 285 W = 48" long and 18 " in diameter=56kg. For 125W mass=(125/285)*56 = 24.6 kg + 10%factor = 28kg. assume power is proportional to volume maintain same diameter and calculate new length = 21" +5" extra.

ENCLOSURE 4

delta v for all cases= 3.57 from MMPC Data modified base on material from ref 1, 10, and 14.

Solar Arrays 20 kg/kw (ref14)
ppu = 5 kg/kw (ref 10)

25 kg/kw Therefore, mass of power unit = 25 P where p= 10 kw for 2 engines.

Engines: 2 ion engines, power harneess, beam neutralizer, gimbals, and support structure = 33.3 kg (ref1), fuel tank factor 0.0335Mp, add one redundant tank of 10 kg, Engine thrust rated at 0.2N/engine (Ref 10). Therefoere the Mass Dry equation results.:

Mprop/dry kg = (34 +0.34Mp +250)N + 10

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MARK R. SCANLAN
PROPULSION
GROUP 4
[redacted]

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4-1

AEROBRAKE

My name is Ed Alcock. I am the aerobraking specialist for group 4. My mission is to allow the spacecraft to obtain a circular orbit around Mars using aerobraking and get the payload (aircraft) to the martian surface.(9) In this design I will talk on 8 main components of my subsystem. They are the heat shield size and shape, the placement of components on the heat shield, dynamics and control of the spacecraft/heat shield, power requirements for aerobraking and landing the payload on the martian surface, communication with the aircraft, separation method, reentry and landing system, and interactions with the other subsystems.

REQUIREMENTS

My subsystem requirements are to:

- use less fuel for more payload room
- dissipate orbit energy in atmosphere
- protect the spacecraft
- perform mass trades
- provide for:
  - aerobrake attitude
  - separation control
  - parachute control
- design effective components for:
  - heat shield structure
  - thermal protection system
  - parachuting/landing

TRADE STUDY

But first, I am going to perform trades and discuss methods of attack.

The first trade involves the method we use to circularize the orbit around Mars from a hyperbolic trajectory from Earth. Contrasting the all-propulsive, aerocapture, and aerobrake methods: (5)

	all-propulsive	aerocapture	aerobrake
technology level	too primitive	may not be available in 1998	8 points available in 90's
amount of fuel	incredible amount for 'retro-launch'	5 points no fuel required	delta-V burns
heat shield mass	none 2 points	~10000 kg	~1000 kg
RFP says:	does not meet rqmt.	does not meet rqmt.	5 points meets rqmts.
score	2	5	8

Aerobrake wins out. Therefore I dissipate orbit energy in atmosphere and use less fuel.

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The second trade concerns the type of shield we are using. Comparing conical and hemispherical heat shields, the conical wins out.

HEAT SHIELDS (5)

	hemispherical	conical (20 degree slope)
wake angle	14 degrees	10 degrees 5 points
surface area / effective area	~2	~1.1 7 points
attitude control	4 points more parallel streamlines	more drag and less control
score	4	12

Therefore, I trimmed down the mass and protected the spacecraft, meeting such a requirement.

The third trade concerns the frontal ceramic tile on the heat shield. Contrasting standard tile with FRCI tile:

CERAMIC TILE (6)

	standard tile	FRCI
cost	less 3 points	more
heat conduction	more but not high	the least 5 points
aerobraking fuel savings	can't go unusually deep into atmosphere	can go deep 6 points
score	3	11

As you can see, FRCI tile wins out. Therefore, I designed an effective component for the heat shield structure and thermal protection systems, thus meeting such requirements.

METHOD OF ATTACK

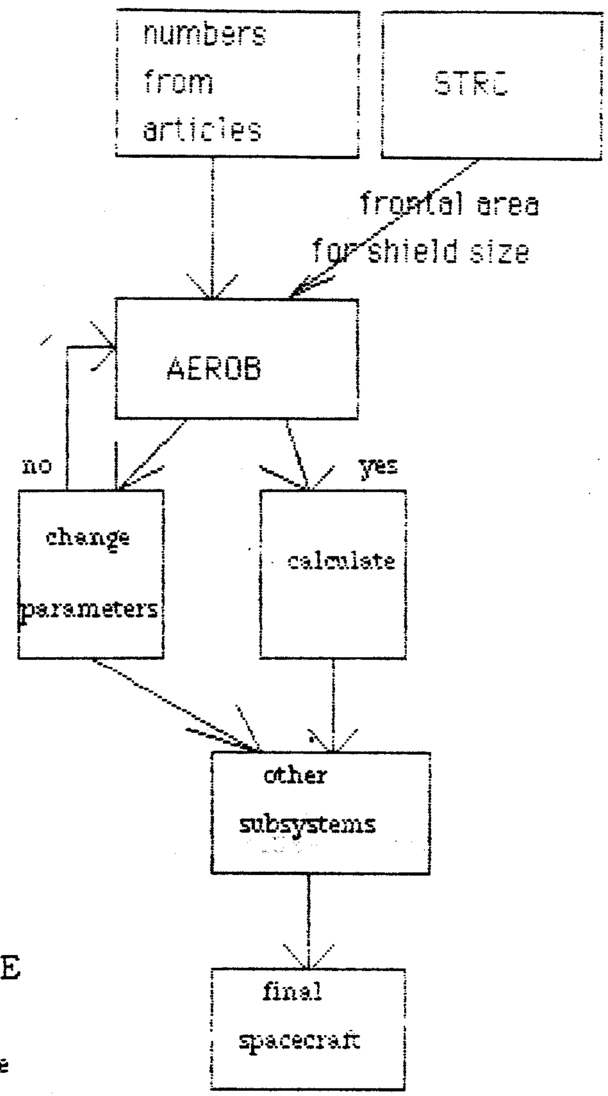
My method of attack starts with the many articles we aerobrakers got from the library search. Reading the articles I got the information to calculate or determine:

-heat shield structure/mass
-ceramic materials
-wake angle information
-temperature information on ceramics
-delta-V specifications
-data on landing payload on martian surface
-"exploding" structures

From STRC I got the spacecraft cross-sectional area and length to allow me to calculate the effective heat shield area.

From there I tried pertinent data into the AEROB program. I change all the numbers in my input file for AEROB until I hit a maximum payload mass for an initial semimajor axis value given to me by MMPC.

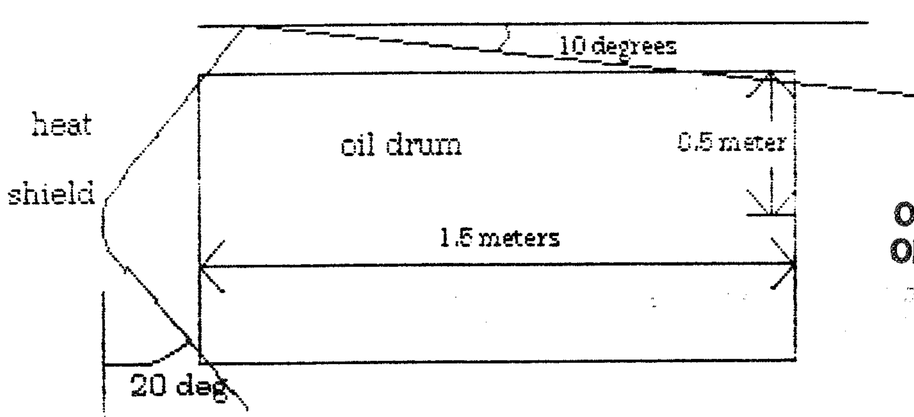
After finding the maximum payload value I print out the AEROB results. From there I gave delta-V values and final spacecraft mass to MMPC and required fuel mass to PPCS. I will talk about the determination of the size and shape of the heat shield.



SIZE/SHAPE OF AEROBRAKE

Considering the conical shield (sloped back 20 degrees) the wake flow angles in 10 degrees behind the shield (14 degrees for a hemispherical shield). (6)

Therefore the size of the shield depends on the geometry of the spacecraft. For example, if I wanted to make a heat shield for an oil drum, length equaling 1.5 meters and radius equaling 0.5 meter,



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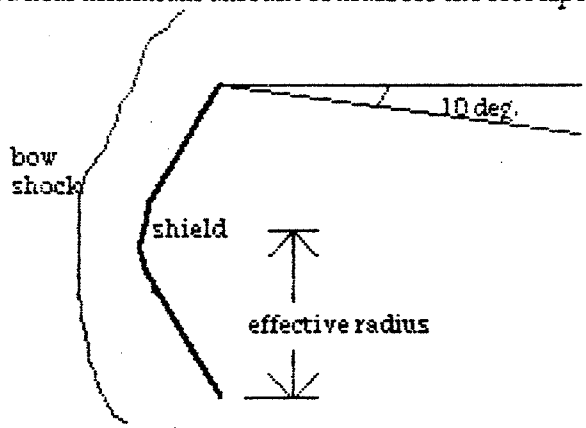
For the heat shield:

effective radius of heat shield = radius of drum + (length of drum)(tan 10 degrees)

and

Effective area of heat shield = (pi) (effective radius of heat shield)**2

The heat shield is sloped back 20 degrees to allow just enough air to flow around the shield while keeping resulting bow shocks away from the spacecraft, a near maximum amount of drag for braking, and a near minimum amount of mass for the corresponding effective area.



Through trading, I chose FRCI ceramic tile, 0.4 centimeter thick, for the frontal surface of the shield. It can take temperatures of up to 740 degrees celsius (900 degrees celsius for 0.5 cm. thick FRCI), enough to cover any orbit insertion. (6)

Behind the FRCI is the ceramic "cloth," much like wire mesh, holding the FRCI tiles together that are "molded" into the cloth.

Behind the shield are 10 "pads" of ceramic insulation, in circles 17 centimeters in radius and 1 centimeter thick.

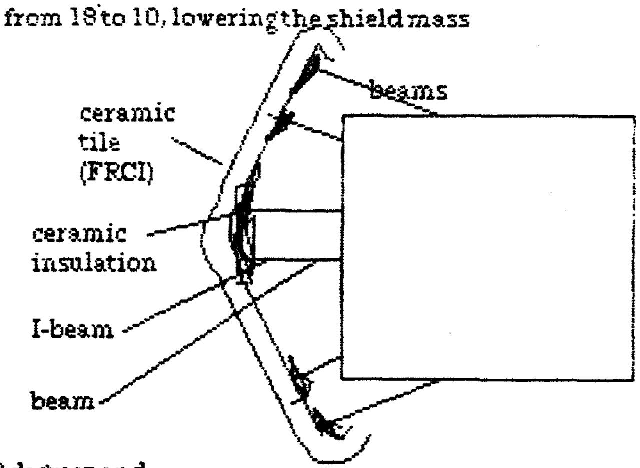
On each pad of insulation goes a cylindrical I-beam, about 2 centimeters in radius and 6 centimeters in length. The I-beams are made of polyimide-graphite, a strong, metallic-like material that conducts very little.

Attached to the I-beams are beams attaching the shield to the payload. Each of the 10 beams (1 per I-beam) is made of aluminum and graphite, as outlined in STRC.

Since such a heat shield design can be applied to larger shields, this shield will easily hold its own. (6)

I determined the shield mass, looking at larger shields and proportioning down, is 980 kilograms. I found a conical shield, 13.4 meters in effective radius, maximum temperature being 900 degrees celsius, with a mass of 6600 kilograms. Since mass is proportional to area, which is proportional to the radius squared, I shrunk down the shield to an area of 120 square meters, a radius of 6.18 meters, to have a mass of 1408.8 kilograms cubed. I lowered the frontal tile thickness to 0.4 centimeter, shrinking the shield mass to 1304 kilograms. Since smaller shields do not need as much of a support structure, I lowered to number of beams, I-beams, and insulation pads from 18 to 10, lowering the shield mass to the present 980 kilograms.

SCIN will place sensors inside the heat shield tile that will be uncovered for sensing upon reentry into the martian atmosphere to send the payload to the surface. Therefore I protect the spacecraft.



DYNAMICS AND CONTROL OF THE HEAT SHIELD

As I stated before, the shield has a wake angle of 10 degrees and a bow shock out in front of the shield. In order to add attitude control to the spacecraft we moved the center of gravity of the spacecraft as close to the shield as we could. Yet the spacecraft can still tumble, therefore needing an attitude and articulation system. AACS talks about this, using an onboard computer to sense trajectory, attitude, and altitude, and side thrusters that thrust cold gas in one or more of 4 directions when needed to correct spacecraft attitude angle at 0 degrees (straight ahead.). The aerobrake attitude requirement has been met sufficiently. (4)

POWER REQUIREMENTS FOR AEROBRAKING AND LANDING PAYLOAD ON SURFACE

Considerable power is needed for aerobraking. Here 3 sets of burns are used: the delta-V required to achieve elliptical orbit around Mars, the numerous raise to periapse burns, and the final burn to circularize the orbit. The mass of fuel needed to perform each of these 3 types of burns is stated in an AEROB printout. But since we could not get an AEROB program with martian data to work for anything (sample printout representing all trials included) MMPC calculated the 3 fuel mass figures using equations from a paper by Dr Stephen J. Hoffman. (1)

For landing the payload on the surface, such parameters for fuel mass are needed:

velocity at beginning of burn = 45 meters per second
acceleration due to gravity on Mars = 3.75 meters per second squared
altitude = 2 kilometers
mass of the payload = 3339 kilograms

MMPC and PPCS calculated all the fuel masses.

COMMUNICATION WITH THE AIRCRAFT

The payload will have an onboard computer, used for detecting payload altitude, attitude, and velocity, and controlling attitude and articulation control, parachute employment, heat shield separation, the onboard thrusters, and the "landing gear". With the aircraft

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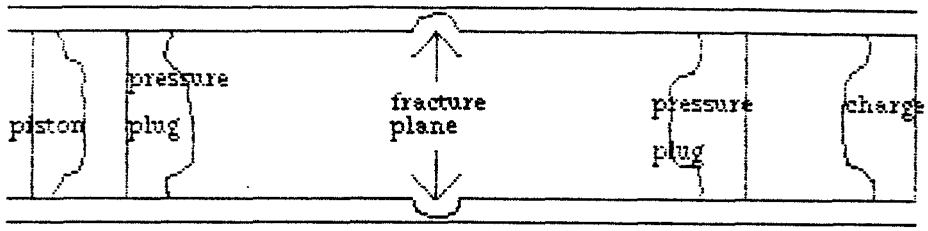
on the surface it can communicate with the bus. Also the aircraft can communicate with the orbiting bus, orbiting around Mars 9 times per 24 hours. CCC takes care of communications between the aircraft and the bus.

SEPARATION METHOD

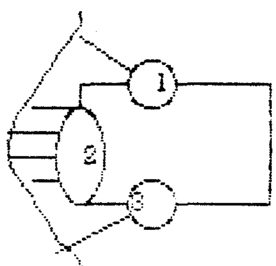
The separation control requirement has been met. Here I will discuss separation between the payload and the bus, the payload and the heat shield, and the payload and the parachute system.

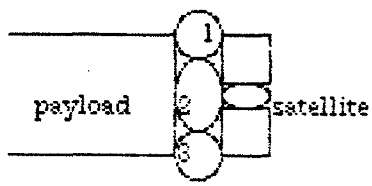
Between the payload and the bus only trusses connect the two. Therefore we will employ "exploding" trusses, employing two pressurization plugs, and a fracture plane per beam. Each exploding location will also employ 1 piston and 1 charge.

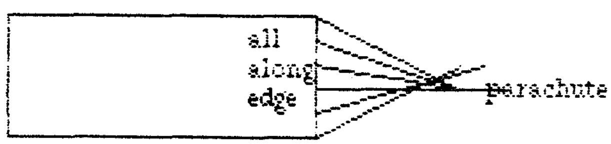
truss, beam, or bolt



With the charge activated, the pressure plugs rupture and the high pressure splits the beam at the fracture plane. The locations of separation on each of the 3 areas mentioned are as follows:







REENTRY AND LANDING CONTROL

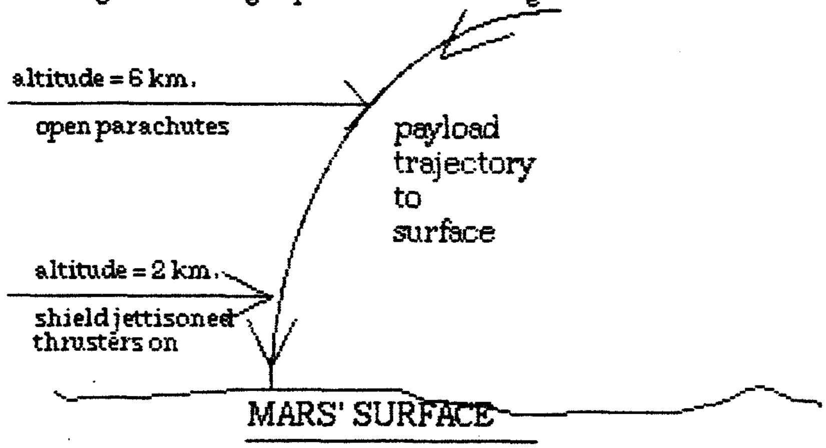
This scenario starts when the payload separates from the bus by the use of "exploding trusses". The payload then is turned along and set on a trajectory on an angle 50 degrees toward the surface from the bus trajectory, set at a speed of 4 kilometers per second. (5)

The payload then dives into the atmosphere at a speed of 4 kilometers per second at an angle of 50 degrees with respect to the martian surface. Attitude control keeps the payload oriented with the velocity.

At an altitude of 6 kilometers and a mach number of 1.2, speed~110 meters per second, the parachute system opens. At an altitude of 2 kilometers, a speed of 45 meters per second, the heat shield is exploded off and the thrusters, located beneath the heat shield, turn on.

The thrusters take the payload to just meters above the surface, where the "landing gear" is activated. The thrusters, holding the payload with a speed of 0+ meters per second, gently place the payload on it's "legs" on the surface.

The onboard computer inside the payload keeps track of the velocity, altitude, and attitude of the incoming payload. It also activates the attitude and articulation control, and performs tasks for SCIN and CCC. My parachuting and landing requirements are being met.



PARACHUTE SIZING

To size the parachutes, I took an example of an actual craft employed previously with parachutes for reentry (i.e. Apollo missions). Typical numbers for Apollo are as follows for Earth's atmosphere (10):

Apollo x: mass ~3700 kg.
diameter of parachutes~18 m.
number of parachutes =3
mass of parachutes=13.2 kg. apiece

Since (diameter of parachute) is proportional to (pressure) is proportional to (mass of payloadd) I converted from Earth atmospheric pressures to martian pressures, from a payload mass of 3700 kg. to one of 5000 kg., keeping 3 parachutes, I came up with for Mars:

payload: mass = 4500 kg.

number of parachutes = 3
radius of parachutes = 49.6 m.
mass of parachutes = 350 kg.

ABOUT AEROB:

In MEB only 3 terminals of all the IBM PC's give results at all, but in terms of Venus. Lembeck's programs do not work, whether one uses a drag coefficient of 0.1, a shield area of 1 square meter, or not. Therefore, we turned to using material from Dr. Stephen J. Hoffman that mentions equations behind the AEROB program. Using this, with correct atmospheric and gravitational data for Mars, we proceeded to calculate the delta-V's and the required fuel masses ourselves. From Hoffman we have: (1)

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$$\text{delta-V} = \frac{-(\text{rho})_p V_p^{2} \; a\, Cd \;\; A\,(1-e)}{m} \left(\frac{2h}{\mu e}\right)^{0.5} \left(\frac{\pi}{4}\right)^{0.5}$$

where each of the variables were explained by Dr. Hoffman in his pamphlet.

As for the required fuel mass for the aerobraking sequence: (1)

$$\text{new mass} = \text{old mass} \quad \exp\left(\frac{\text{velocity at periapse}}{\text{(specific impulse)(g)}}\right)$$

MMFC did the mass and delta-V calculations.

COSTING OF THE AEROBRAKE STRUCTURES

For the aerobrake subsystem, the total mass, including the heat shield, heat shield structure parachutes, and parachute covers, equals 980 kg. plus 418 kg., which equals 1398 kg..

I want the heat shield to be new. Therefore, the total number of labor hours put into the shield are as follows: (2)

$$\text{\# of hours} = 8481.0 \text{ hours (mass of aerobrake)}^{0.8416}$$
$$= 1145829.2 \text{ hours}$$

I want the parachute system to be used. The total number of hours put into the parachutes are:

$$\text{\# of hours} = 4662.0 \text{ hours (mass of parachutes)}^{0.5}$$
$$= 95314.8 \text{ hours}$$

Therefore, the total number of labor hours is the two figures added together, which is 1241144.0 total hours.

Since a typical engineer gets paid $10 per hour, the total cost of the aerobrake structures is:

($10 per hour)(1241144.0 hours) = $ 12411440.0

PARACHUTE COVERS

The average thickness of a parachute skirt is 0.5 cm. (7) Therefore, the volume of the material of one parachute is:

volume = (1.5) (effective area) (thickness of material)

= (1.5)(pi) (49.6 m.)**2 (5*10**-3 m.)

= 57.996 meters cubed

The thickness of 1 parachute on top of 1 of 3 payload cylinders is:

thickness = (volume of parachute)/ (area of payload cylinder top)

=57.996m**3/(pi)/(4.6**2 m**2) = 0.87 m

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We then encase each parachute, folded and mashed down under pressure under aluminum covers, 1 for each of the 3 payload cylinders.

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G4
AERO

Lemke's AEROB program
a:oof.dat

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AEROBRAKE SHIELD AREA = 120.00 M2

ORBIT	TIME (DAYS)	SCMASS (KG)	DELTA M (KG)	RP (KM)	VP (KM/S)	DV (M/S)	A (KM)	LCE
0	.00	10100.00		3500.0			250000.0	.9860
0	21.91	7954.94	2145.06	3500.0	4.942	704.7	250000.0	.9860

NO JETTISON OF STAGE INERTS WHICH REQUIRED 214.5 KG

```
 40
.210811E+02     .203942E+02     .500000E+00     .500000E+00
.329061E+03     .330516E+03     .354619E+03     .354619E+03
 1       4      2
 41
.210811E+02     .203942E+02     .500000E+00     .500000E+00
.329061E+03     .330516E+03     .354619E+03     .354619E+03
 1       4      2
 42
.210811E+02     .203942E+02     .500000E+00     .500000E+00
.329061E+03     .330516E+03     .354619E+03     .354619E+03
 1       4      2
 43
.210811E+02     .203942E+02     .500000E+00     .500000E+00
.329061E+03     .330516E+03     .354619E+03     .354619E+03
 1       4      2
 44
.210811E+02     .203942E+02     .500000E+00     .500000E+00
.329061E+03     .330516E+03     .354619E+03     .354619E+03
 1       4      2
 45
.210811E+02     .203942E+02     .500000E+00     .500000E+00
.329061E+03     .330516E+03     .354619E+03     .354619E+03
 1       4      2
 46
.210811E+02     .203942E+02     .500000E+00     .500000E+00
.329061E+03     .330516E+03     .354619E+03     .354619E+03
 1       4      2
 47
.210811E+02     .203942E+02     .500000E+00     .500000E+00
.329061E+03     .330516E+03     .354619E+03     .354619E+03
 1       4      2
 48
.210811E+02     .203942E+02     .500000E+00     .500000E+00
.329061E+03     .330516E+03     .354619E+03     .354619E+03
 1       4      2
 49
.210811E+02     .203942E+02     .500000E+00     .500000E+00
.329061E+03     .330516E+03     .354619E+03     .354619E+03
 1       4      2
 50
.210811E+02     .203942E+02     .500000E+00     .500000E+00
.329061E+03     .330516E+03     .354619E+03     .354619E+03
```

Attitude and Articulation Control System

for the group 4

Aries Spacecraft

Tim Hogan

Environmental Effects on Satellite Motion

The only data obtainable about external forces around mars was the atmospheric data in the table shown. The results have been extrapolated to approximate the atmosphere at orbital altitude.(1400km.) The estimated density at this altitude is, d =10^{-44} kg/m^2.

area = 5 m^2 (approximately)

orbital period = 9600 seconds,

orbital circumference = 29,670,000 m =c,

mass encountered/time = d*A*c/t = 3.09063 x 10-41seconds

= 9.75 x 10-33kg/10 years

This is less than one molecule of uranium in one million years. Therefore the aerodynamic forces will be neglected.

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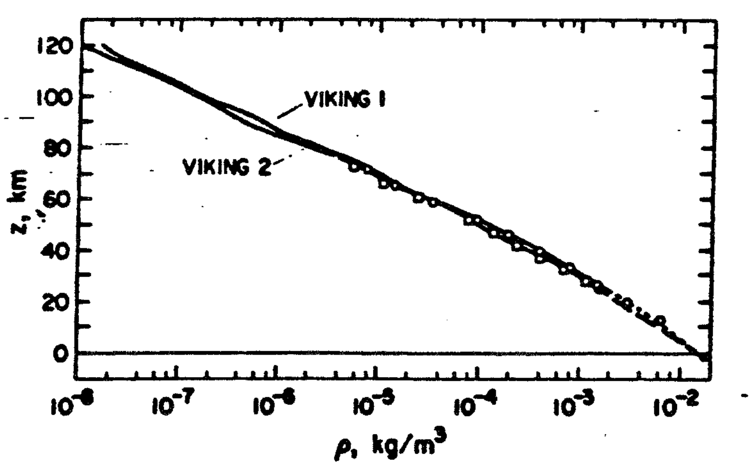

Fig. 10. Density profiles of Mars atmosphere to 120 km. Data shown by curved lines above 28 km are derived from accelerations. Points are from stagnation pressures (circles, Viking 1; squares, Viking 2). Densities from measured pressure and temperature during parachute descent are shown in the lowest 5 km.

Effects of tape recorders on angular velocity of satellite

If a tape recorder is turned on, the angular momentum of the s/c will be conserved.

w_1 = angular velocity of tape recorder reels.

w_2 = angular velocity of s/c.

I_1 = moment of inertia of tape reels.

I_2 = moment of inertia of s/c about axis of rotation of tape reel, less the moment of inertia of the tape reel.

$$(Iw_1 + Iw_2)_{\text{before spinning}} = (Iw_1 + Iw_2)_{\text{after spinning}}$$

$I_1 = .002\ m^2kg$

$I_2 = 850.5\ m^2kg$

Before spinning:

$w_{1i} = 0.0$

$w_{2i} = 0.0$

After spinning:

$w_{1f} = 3.14$ rad/sec

w2f= 7.3876 x 10-6 rads/sec

w_r = angular velocity of limit cycle = 6.5 x 10-6 rads/sec

Since this is less than twice the angular velocity of the limit cycle, a maximum of two pulses would be required to bring the angular velocity of the s/c within desired limits. If two pulses turn out to be excessive in fuel consumption, or the angular momentum of the mass storage device turns out to be higher than calculated here, then the logic of the s/c may control when the tape may be turned on. It could then turn the tape recorder on when w_{1f} is of the same sign as w_r of the s/c about the axis parallel to the tape. Since this would impart an angular acceleration to the s/c opposite to the direction in which it is turning, this would tend to cancel the limit cycle angular velocity instead of making it farther outside the desired limits.

Basic Layout of Satellite

Control system needs had a large impact on overall satellite configuration. The original proposed configuration had an axis other than the maximum moment of inertia about the pitch axis of the s/c, with the s/c rotating about the pitch axis once per orbit as shown in the diagram. This is somewhat unstable due to energy dissipation, so some of the appendages were moved. An alternative would have been to have a momentum wheel spinning in the opposite direction with an equal angular momentum in the opposite direction. thus cancelling it out.

The antenna and platform pointing are accomplished with the use of precision gimbals which track at below limit cycle angular velocities, so that they may compensate for any small movements during the limit cycle.

The antenna is on the side of the spacecraft away from the planet, the reason being that the earth is in the general direction of the sun from Mars, so the "back" of the spacecraft faces in the general direction of earth during the day, while imaging . The gimbals can keep the antenna pointed toward earth most of the day while imaging, and the tape recorders can tape the information if the earth is obstructed.

The actuation method chosen was thrusters, since

1) Thrusters would be needed anyway

2) The mass of the propellant plus thrusters was not excessive

3) The thrusters would provide excellent accuracy

The pointing of the imaging equipment was chosen to be done with precision gimbals because:

1) accuracy requirements

2) The need to have the thrusters off during imaging to prevent fouling the lenses

The gimballing system used was chosen over an inertial platform which rotated with respect to the rest of the satellite because of:

1) simplicity

2) reliability

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Thruster placement on satellite

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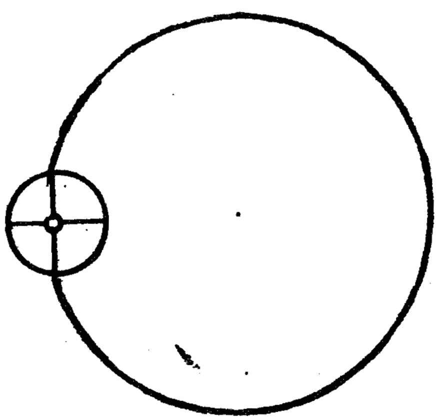
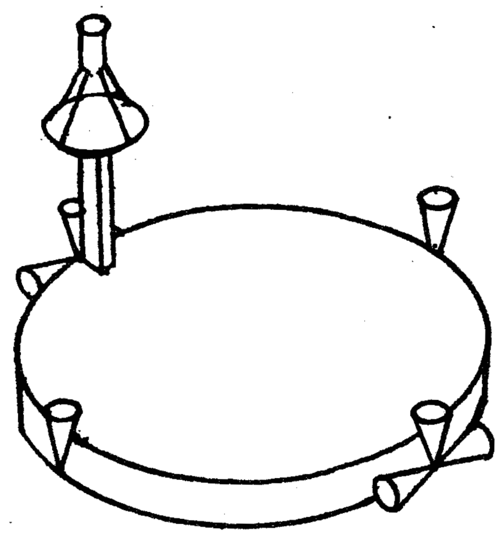
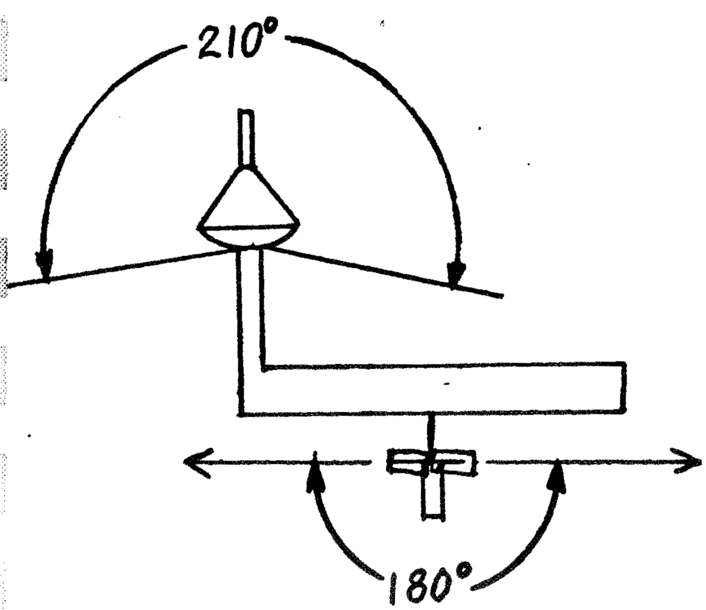
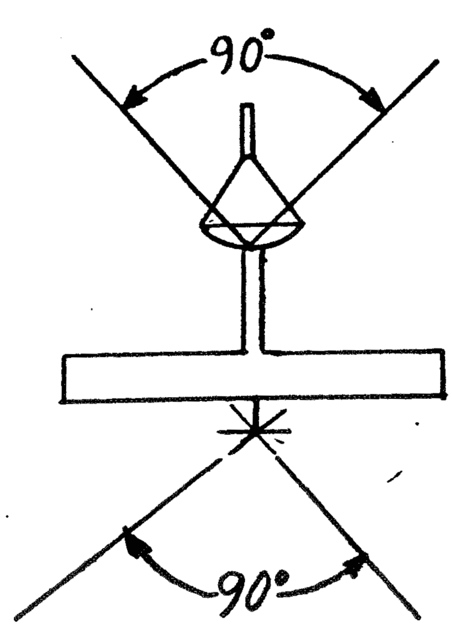
The thruster placement on the satellite is shown in the diagram. The thruster placement used on the satellite gives both large enough maneuvering torques, and small enough impulses to give fine control. The placement shown gives as large a torque arm as possible without building additional structure, which would either be excessively large,heavy, or would be excessively flexible if made light, and small. The thruster layout shown gives rotation about all axes, and translation in one direction. If small stationkeeping adjustments are required the spacecraft may be reoriented temporarily for the thruster fire. It does the control with sixteen thrusters, eight of them redundant, so there is no single point failure. None of the thrusters face in the direction of the imaging equipment.









Thruster Placement on Payload Vehicle

The following constraints were placed on thruster placement
1) To save weight the thrusters must be placed so that no additional structure need be created specifically for them.

2) The thrusters must not point away from the aerobrake, because that would blast toward the earth-mars booster and satellite during transit, which would not only foul the spacecraft with residue, but would negate some of the momentum transfer.

3) The thrusters must not point toward the aerobrake for the same reasons stated in (2) above.

4) The thrusters must provide rotation about all axes, but need not provide translation, the translation being provided by the main engines.

5) The thrusters must provide rotation about the vertical axis during the retrofire and landing, due to the fact that the engines are thrust varying and not gimballed.

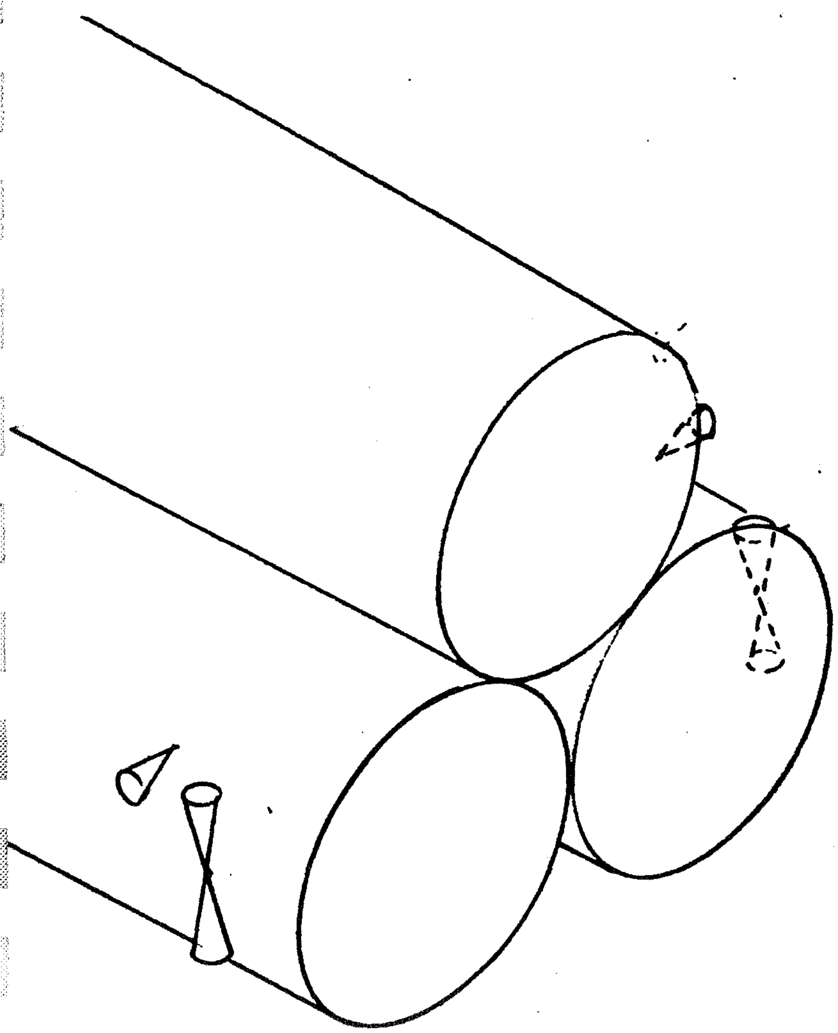
6) The thrusters should be placed with as long a moment arm as possible to minimize fuel consumption.

7) The thrusters must provide control during aerobraking and landing as well as during the earth-mars leg of the mission.

8) The thrusters should be close to to the fuel tank, to save the mass of longer fuel lines.

The configuration shown on the next page satisfies these requirements best.

Payload Landing Vehicle Thruster Placement



Thruster sizing

These calculations are used to find the thruster sizes to perform a maneuver given by the instructor to be performed once per day. The assigned maneuver is to rotate the spacecraft at a rate of one complete rotation in 360 seconds. The acceleration specified is taken from reference two.

a = angular acceleration
r = torque arm of thruster(s)
F = total force exerted by thruster(s)
T = torque
I = moment of inertia about axis in question
t = time in seconds to accelerate to desired angular velocity
w = angular velocity required for manuever = .01745 rads/sec
t = w/a
a = T/I = 10^{-3} rad/sec (ref 5)
T = r*F
giving F= I*a/r

for satellite: I = 850.5 kgm^2, r = 4.4 m, t = 17.5 s, F = .190 N

Using these results, two .5 N thrusters were chosen for each thruster pod. Using two thrusters in unison gives more than the chosen angular velocity. If possible, thrusters as small as .1 N or .2 N would be used to lower limit cycle angular velocity, and increase limit cycle time. However no data was available on whether or not thrusters of such size are available or satisfy durability requirements. Also, two thrusters would have to fail before s/c becomes uncontrollable.

for payload: I = 110,000 kgm^2, r = 5 m, t = 17.5 s, F = 21.98 N

Using these results, two twenty N thrusters were chosen for the payload. These thrusters are also sufficient to use for the combined confuguration as the thrust required is met. This also prevents catastrophic control loss in the event of one thruster failing.

Deadband calculations (Limit cycle)

Using smaller thrusters gives smaller residual angular velocity, thereby increasing the pointing accuracy possible. Smaller residual velocity decreases limit cycle propellant usage, and allows a greater time for the limit cycle. Increasing limit cycle time gives more time for vibrations from thrusting to dampen out, and more time for the thruster gas to disperse. For imaging equipment and other sensitive instruments, thruster gas can cause the optics or other sensors to be coated with residue, reducing performance of, or ending the useful life of the instruments.(ref 3)

Limit cycle analysis

Smallest deadband obtainable is limited by the smallest impulse which can be generated by the thruster. The impulse varies with the time of the pulse. This time is limited by the speed of the microprocessor, and is .005 seconds.(ref 4) Impulse is also limited by the smallest torque which can be generated. This smallest impulse gives a residual angular velocity I will call w_r.

$w_r = r*F*t/I$

$t = .005$ sec

For Satellite:

$F = .5$ N

$I = 850.5 m^2 kg$

$r = 2.2$ m

$w_r s = 6.5 \times 10-6$ rads/sec

dws15 = 5.8 x 10-3 rads

TABLE I. EPS CONTROL SYSTEM REQUIREMENTS

System Axis	Command Pointing Uncertainty	Stability for 15 Min
EPS X (pitch)	$\pm 1.1 \times 10^{-5}$ rad	$\pm 1.1 \times 10^{-5}$ rad
EPS Y (yaw)	$\pm 1.1 \times 10^{-5}$ rad	$\pm 1.1 \times 10^{-5}$ rad
EPS Z (roll)	$\pm 2.9 \times 10^{-3}$ rad	CMG control system

TABLE II. CMG CONTROL SYSTEM REQUIREMENTS

System Axis	Command Pointing Uncertainty	Stability for 15 Min
CMG X (pitch)	$\pm 1.2 \times 10^{-3}$ rad	$\pm 2.6 \times 10^{-3}$ rad
CMG Y (yaw)	$\pm 1.2 \times 10^{-3}$ rad	$\pm 2.6 \times 10^{-3}$ rad
CMG Z (roll)	$\pm 2.9 \times 10^{-3}$ rad	$\pm 2.2 \times 10^{-3}$ rad

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Change in angle in 15 minutes (dw15) with no thrust applied is less than the error of skylab (shown in the table on previous page). Since skylab used it's EPS (experiment pointing system) for it's fine pointing, it seems a very logical type of system to use. Analysis of the design supports the ability of the design to satisfy pointing requirements. Since the change in attitude is very small in 15 minutes, although it is three times greater than than the pointing uncertainty of skylabs CMG systems. But since the angle which can be adjusted for by the gimbals is much larger than dw15, the thrusters could be left off for quite some time, if necessary. The speed of gimbals is typically much higher than this.(ten to one hundred times greater). This time could be used to dampen vibrations, and/or let the propellant gases dissipate before opening lens caps, for instance.

Another alternative for the satellite would be to have very fine vernier cold gas thrusters to reduce the limit cycle angular velocity. CMG's could not be used on the satellite because the spacecraft is rotating constantly. Even if this were not the case, cmg's would not be advisable because of their complex control laws, cost, and the fact that they are not needed.

for payload landing vehicle

$F = 10\ N$

$I = 110{,}000\ m^2kg$

$r = 5\ m$

$w_{rp} = 2.3 \times 10^{-6}$ rads/sec

$dwp_{15} \approx 2.0 \times 10^{-3}$ rads/sec

Since the payload has less stringent pointing requirements, this is a satisfactory limit cycle. The 15 minute drift angle is comparable to that of skylab, even though the pointing accuracy of the payload landing vehicle need not be as great. If it is found that more torque is needed for landing roll correction, then the thruster size could easily be increased. (Note: all limit cycles calculated with smallest thrust available about a given axis. For instance, the limit cycles for the payload landing vehicle are calculated using one ten newton thruster for control about one axis.)

Propellant Usage Equations

Since the moment of inertia of the spacecraft is not changing significantly, the equation, m = n(I*dw)/(r*dv) is used.

where:

dw = change in angular velocity required for maneuver.(.01745 rad/sec)

dv = exit velocity of propellant = 2156 m/s

m = mass of propellant ejected

r = moment arm

n = number of times angular velocity is changed

n = (2 pulses/manever)*365 maneuvers/year = 730 pulses/year

for satellite:

r = 2.2 m

I = 850.5 m^2kg, giving mass of propellant = 2.3 kg/year

for payload landing vehicle

r = 7 m

I = 110,000 m^2kg, giving mass of propellant = 61.90 kg/year

Thruster durability

If a 15 minute limit cycle can give the required accuracv, then the number of pulses due to limit cycle thrusting will be 68,000 pulses per year. This stretches the limits of some thrusters but others are rated at up to 500,000 pulses. There is also a spare thruster for each. The long time in space is also an important factor.

Placement and use of sensors

The star trackers are used to find the inertial orientation of the spacecraft. During orbit around mars, the forward facing horizon sensor determines the attitude with respect to the mars local vertical. The star tracker aligned with the pitch axis, focuses on a star near the orbital axis, which is also the axis of rotation of the spacecraft.

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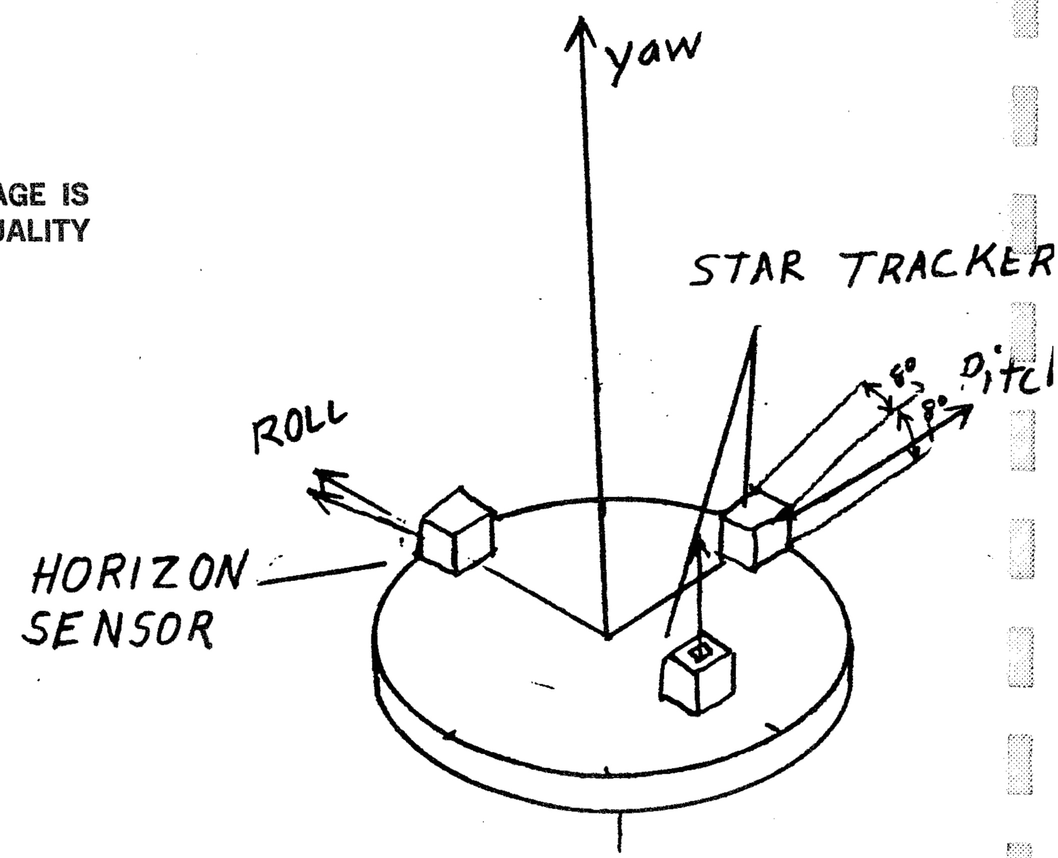

Decoupling of accelerometers from the rotation of the spacecraft.

The accelerometers are not placed at the center of mass of the s/c because of structural design. The accelerometers experience accelerations due to rotation which may be subtracted from the accelerations measured by the accelerometer.

Acceleration due to rotation about the center of mass, of a point (X,Y,Z), due to rotation.

$a = r \times (r \times w)$ (ref2)

$a_x = w_y(w_xY-w_yX) - w_z(w_zX-w_xZ) + d/dt(w_y)^*Z - d/dt(w_z)^*Y$

$a_y = w_z(w_yZ-w_zY) - w_x(w_xY-w_yX) + d/dt(w_z)^*X - d/dt(w_x)^*z$

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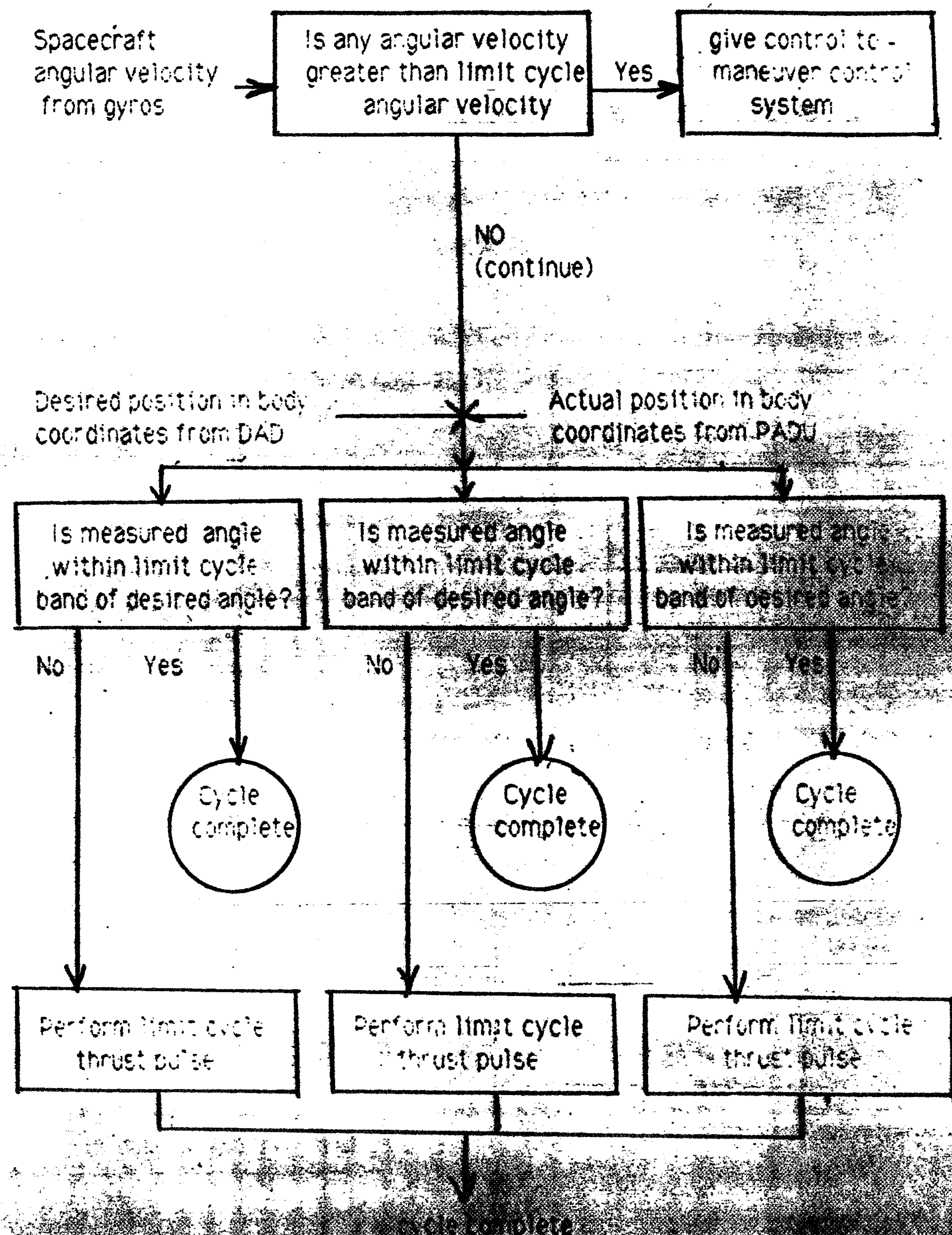
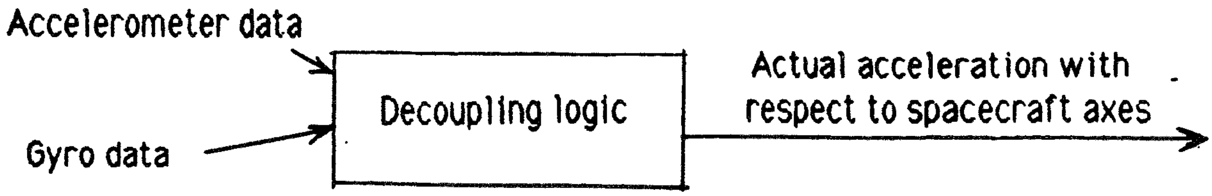
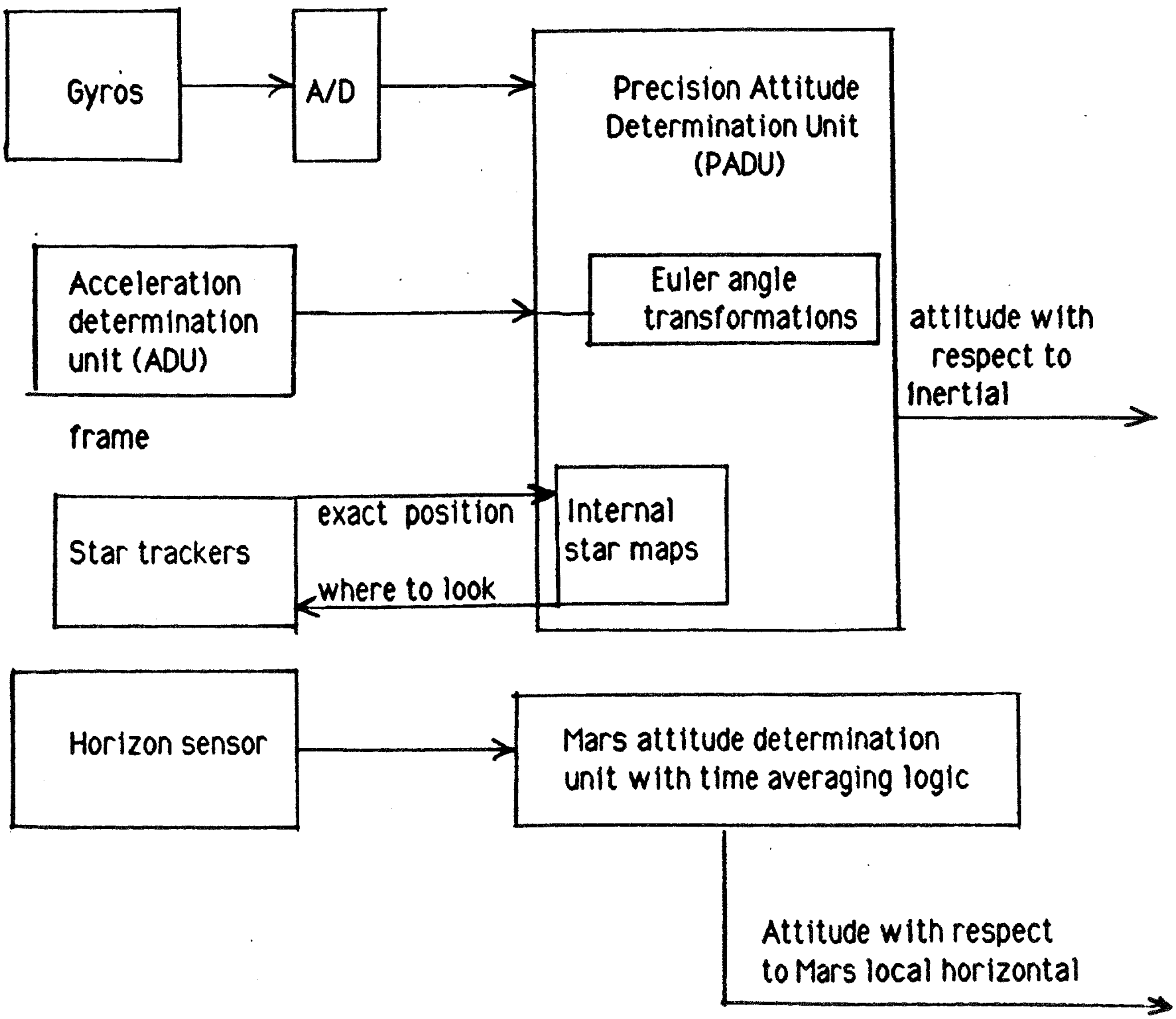
Limit Cycle Thruster Control Unit



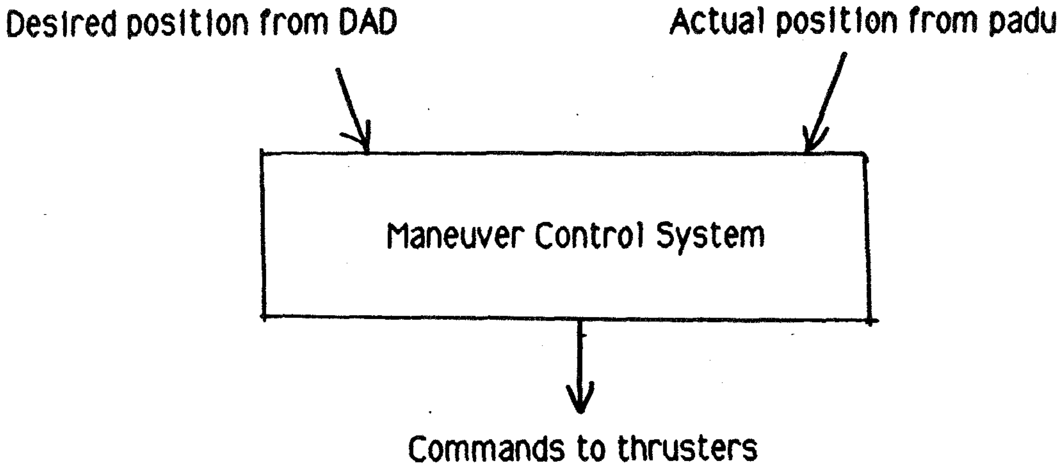
Acceleration Determination Unit



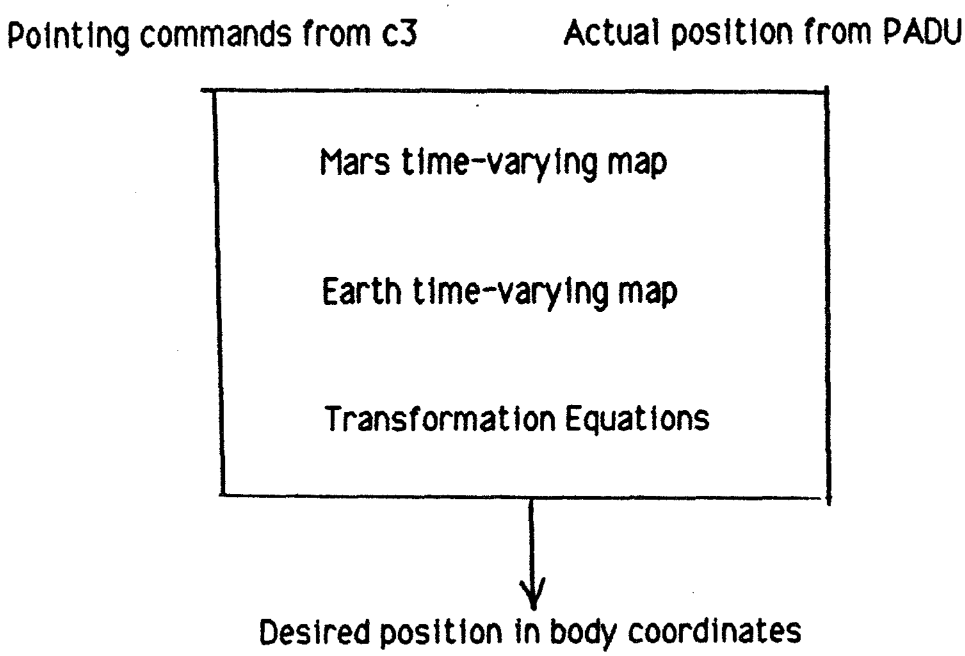
Precision Attitude Determination Unit (PADU)



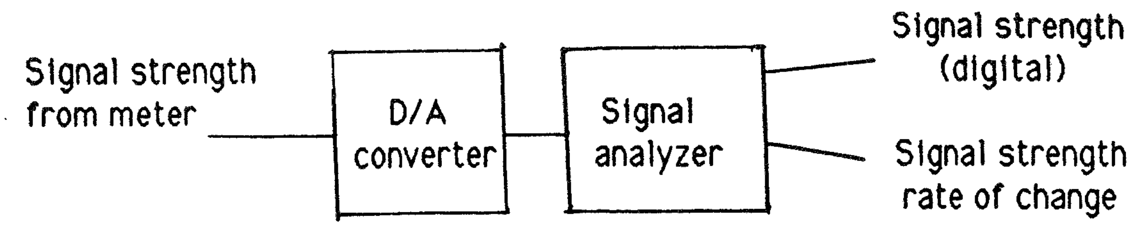
Maneuver Control System



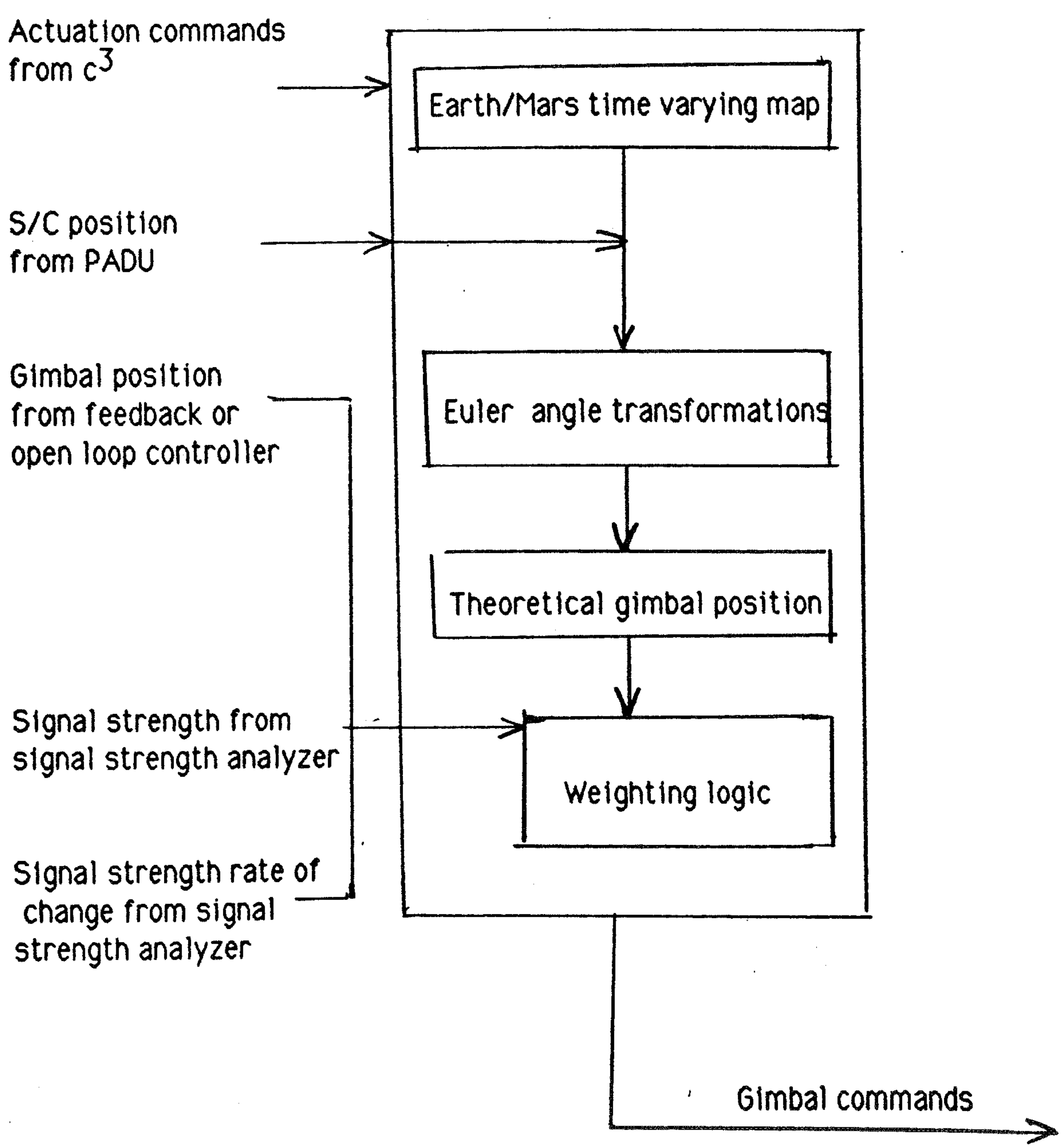
Desired Attitude Determination (DAD)



Signal Strength Analyzer



ANTENNA CONTROL UNIT



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[4]Lembeck, M., Instructor, "AAE241, Aerospace Design Laboratory", University of Illinois, 1988.

[5]Chobotov, V. A., " Spacecraft Attitude Dynamics and Control", lecture notes for AAE 391E, University of Illinois, Spring, 1988.

[6]Seif A., and Kirk, D. B., "Structure of the Atmosphere of Mars in Summer at Mid Latitudes", Journal of Geophysical Research, Sept. 30, 1977

Attitude Control and Articulation Devices on the Satellite

COMMAND, CONTROL, AND COMMUNICATIONS SUBSYSTEM (CCC)

ERIK GUNTER
DESIGN ANALYST

CCC REQUIREMENTS

The requirements of the communications subsytem are as follows:

(1) collect telemetry from the subsystems and insure that engineering and scientific data is transmitted to Earth

(2) receive commands from the Earth (or Mars Station) and send commands to subsystems

(3) control power switching

(4) receive power from the Power and Propulsion Subsystem

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DETERMINATION OF COMMUNICATION FREQUENCY

One of the first decisions which must be made in the CCC subsystem is to determine the method of communication. The frequency range to be used and the mode of operation (direct link to Earth or a series of relay satellites) must be determined. Selecting the mode of operation is the easier of the two tasks. A satellite relay system would involve the construction of additional satellites (assuming that any existing satellites cannot be employed for the mission), but would reduce communication blockage time. A direct link would utilize the Deep Space Network only, and would provide adequate communications for the mission despite increased blockage time (approximately 24% of orbiting time).[1] Therefore, a direct link using existing systems is the more economical choice.

Determining the communication frequency is a more difficult task. The choice involves many different parameters: data transmission rate, antenna gain, immunity to jamming, pointing accuracy, availability, and lifetime. A trade study was conducted (see figure 1), and the X BAND was chosen, mainly due to its pointing accuracy requirement and current availability. Its only drawback was its vulnerability to jamming, which should not be a significant factor for a Mars mission. Also, since this mission does not require the transmission of large quantities of data compared with past exploratory missions, the larger data rates of the other frequencies are not significant.

ANTENNA SELECTION

Earth Communication Antenna

Having determined the mode of communications to be used, the next step is to size the antennas. The requirement of the spacecraft-to-Earth antenna is to fulfill the data transmission requirements of the science subsystem at a distance determined by the mission planner. If all the science instruments on the bus are operating at once, then data can be collected at a rate of 66,000 bits/sec. At Mars encounter, the communicating distance will be approximately 1.05 AU (1.6×10^9 km).

The following assumptions were made for the sizing calculations:

frequency (f) = 8.414 GHz (x band downlink)
efficiency (z) = 0.65 (expected by 1995)
distance (D) = 2.0×10^8 km
DSN antenna (Dr) = 70 m (predicted by 1990)
signal received (Pr) = 1.0×10^{-16} Watts

Using the sizing equation for a directional antenna and solving for the two desired parameters, antenna size and transmitting power:

$$Pr = \frac{Pt}{\left(\frac{4CD}{f\ z\ Dr\ Dt\ pi}\right)^2}$$

The requirements are met by selecting 5 Watts for the transmitting power (Pt) and 1.0 meters for the antenna diameter (Dt). With these selections, the power received at Earth is $Pr = 1.26 \times 10^{-16}$ Watts, which is significantly strong enough for DSN's reception (Voyager's signal strength was 10^{-18} Watts at Jupiter)[2] Using a bandwidth of 15,000 Hz, Shannon's Law gives a data rate of

$$B = W \log(SNR + 1),$$

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where SNR (signal to noise ratio) is 30.43,

$$B = 74{,}600 \text{ bits/sec}$$

This selection more than satisfies the requirements, and also allows additional instruments to be placed on the orbiting bus.

UHF Antenna

A UHF antenna is required to communicate between the bus and the Mars base or to support the operation of the Mars airplane. This antenna will be sized using the formula for an isotropic antenna:

$$Pr = \frac{z \cdot A \cdot Pt}{4 \cdot pi \cdot D^2}$$

Where z = 0.65 and D = 1500 km (orbiting altitude after the payload is separated). It is assumed that the Mars base's antenna can receive a signal of strength $Pr = 10^{-14}$ Watts. The formula reduces to the equation:

$$0.4350 = A \times Pt$$

With this formula, the antenna parameters of Pt = 5 Watts and Area = 0.139 m^2 (0.42 m antenna). With these selections, the maximum communications distance is 1890 km. The orbiting bus will have an altitude of 1380 km, so communication time between the bus and the base is 16 minutes. Once the antenna sizes have been computed, the antennas can be placed on the spacecraft.

ANTENNA PLACEMENT

Positioning the antenna presents another problem: the directional antenna must be pointed at the Earth with a high degree of accuracy for communications. Several options for placement were considered, and the best choice was to place the antenna on a boom below the spacecraft bus (see Figure 2). This location was chosen to reduce the amount of spacecraft rotation used to point the antenna. With this placement, the antenna is limited by the spacecraft to 150^0 of arc up and down. After the payload is separated, the arc improves to 180^0. This constraint is a problem for Earth communications, (data cannot be taken and sent to Earth simultaneously), but adequate communications are available with small spacecraft rotations.

Thrusters will be used by the articulation subsystem, and a pointing gimbal will make smaller adjustments in the antenna's position. The gimbal can manipulate the antenna 75 degrees above or below the horizontal(or 90 degrees after separation), and 45 degrees right or left of center. The gimbal can move the antenna within one second of arc, and the X BAND requires an accuracy of 2 - 3 minutes of arc[3]. Therefore, the pointing requirements should be filled with this arrangement.

Since the UHF antenna is omnidirectional, pointing is not a problem. The UHF antenna will be secured to the underside of the orbiting bus (see figure 2). The UHF antenna can be used to transmit data to the Mars base while the science instruments are operating.

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The computer selected for this mission will have to perform a variety of duties, including sending commands to other subsystems, power switching, and formatting data. Assuming that a computer with 64 K RAM and 8 K PROM will be adequate, then a computer similar to the one designed for the Mars Observer will be used.[4] To accommodate the requirements of the scientific instruments on the payload, a second computer will be placed on the inbound. This computer will not add significant weight or power required (only and additional 1 kg and 1 W), and it will eliminate the need for the payload and the bus to communicate during the descent. This second computer could also be used as a backup system before the bus and payload seperate.

The command system will be a typical command system with the computer sending and receiving signals (see figure 3). Three tape recorders -- one on the orbiting bus and two on the payload -- will be employed to store data from the instruments. The tape recorder on the bus will be used because the antenna position on the bus prevents Earth communications while the instruments are taking data. The two tape recorders (one is redundant) on the payload will store the data from its instruments. When the Mars airplane is retrieved, these recorders will be removed and taken to the Mars base for transmission to Earth (direct link or via the orbiting bus). The orbiting bus's recorder will be similar to the Mars Observer tape recorder, with approximately 1.5×10^9 bits of storage available.[5] With this storage capacity, the instruments on the bus can operate and store data continuously for 6.3 hours. The two recorders on the payload will not need such capacity, since less than 10^6 bits of data will be taken, so a smaller recorder will be chosen for the payload.

DESIGN PROBLEMS

At this time there are still some problems with this subsystem's design. One of the more significant problems is that the components may be susceptible to radiation damage. Radiation exposure may affect the design of the antennas, computers, and tape recorders. Another possible problem is that the payload's componets must be able to withstand the high temperatures of aerobraking. Potential system failures are another problem. The payload's second tape recorder and the second computer are the only backup systems at this time. Another problem is that the antenna has been designed to provide its data rate while the Earth and Mars are nearby (1.25 AU away), and this is only a fraction of the time. Antenna pointing may also be a problem. The spacecraft cannot be used as a continuous link between the Mars base and the Earth because the spacecraft must be rotated for the antenna to "see" the Earth. These problems and others must be resolved before the design is completed.

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[1]White, Ronald E. "Manned Mars Mission Communication and Data Management Systems." Marshall Space Flight Center, AL. p 890.

[2]AAE 241 Class Notes

[3]White, p 895.

[4]The Mars Observer Investigation Description, April 1987.

[5]Ibid.

FREQUENCY SELECTION TRADE STUDY

SELECTION

	S/X BAND	KA/MM-WAVE	OPTICAL
BANDWIDTH	FEW MBPS	INCREASED	MUCH INCREASED
ANTENNA GAIN	REFERENCE	12dB MORE THAN X-BAND	60dB MORE THAN KA
IMMUNITY TO JAMMING	POOR	BETTER	EXCELLENT
SIGNAL ACQ.	EASY	SATISFACTORY	DIFFICULT
LIFETIME	OK	OK	SHORT LIFE
COMPATIBILITY WITH EXISTING SYSTEMS	YES	NO	NO
TECHNOLOGY STATUS	MATURE	IMMATURE (TECH PLANNED)	IMMATURE (SOME RISK)

REFERENCED FROM R WHITE, MARSHALL SPACE FLIGHT CENTER

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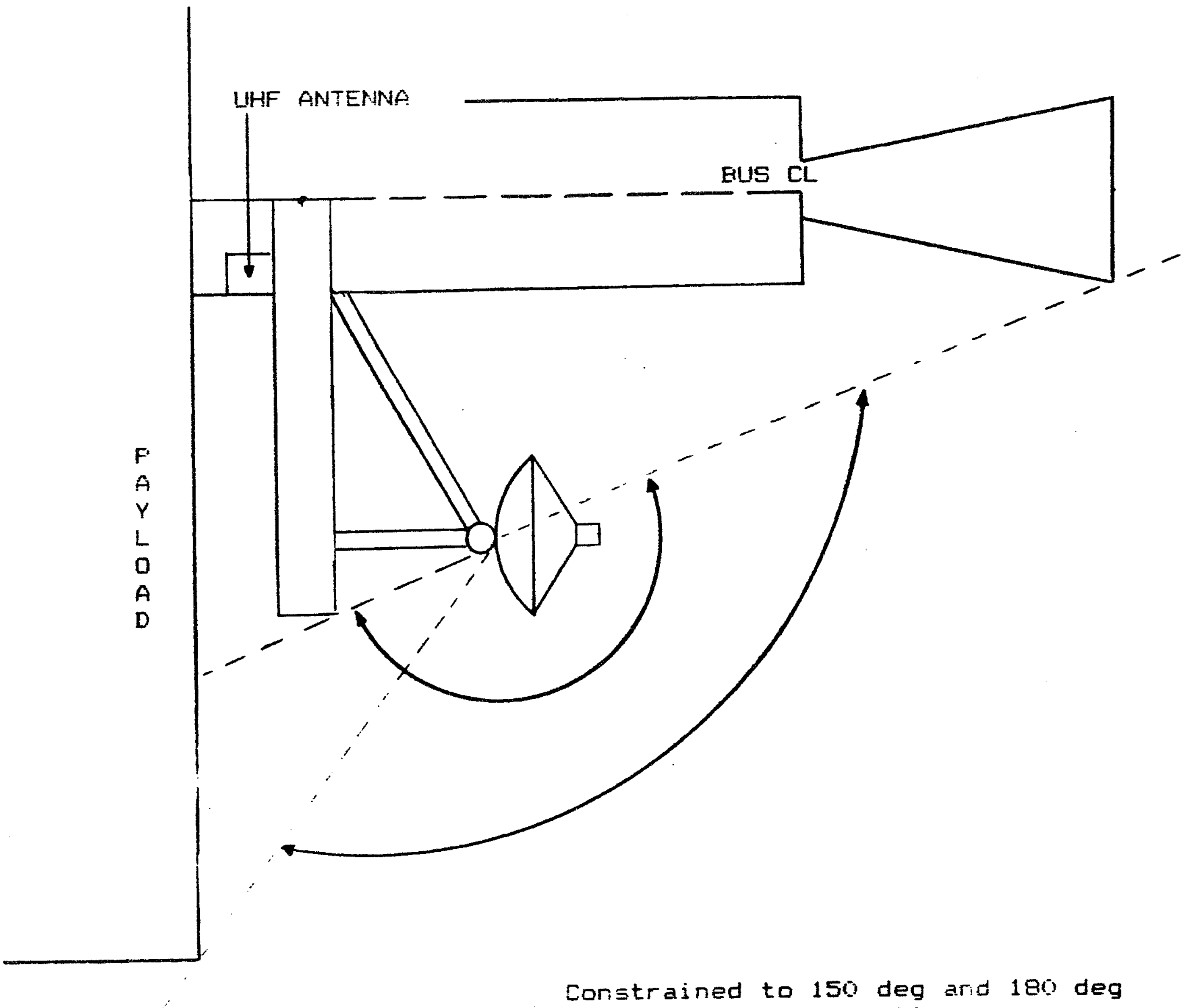
FIGURE 2 ANTENNA PLACEMENT



Constrained to 150 deg and 180 deg
after payload separation

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FIGURE 3 SPACECRAFT CONTROL SYSTEM

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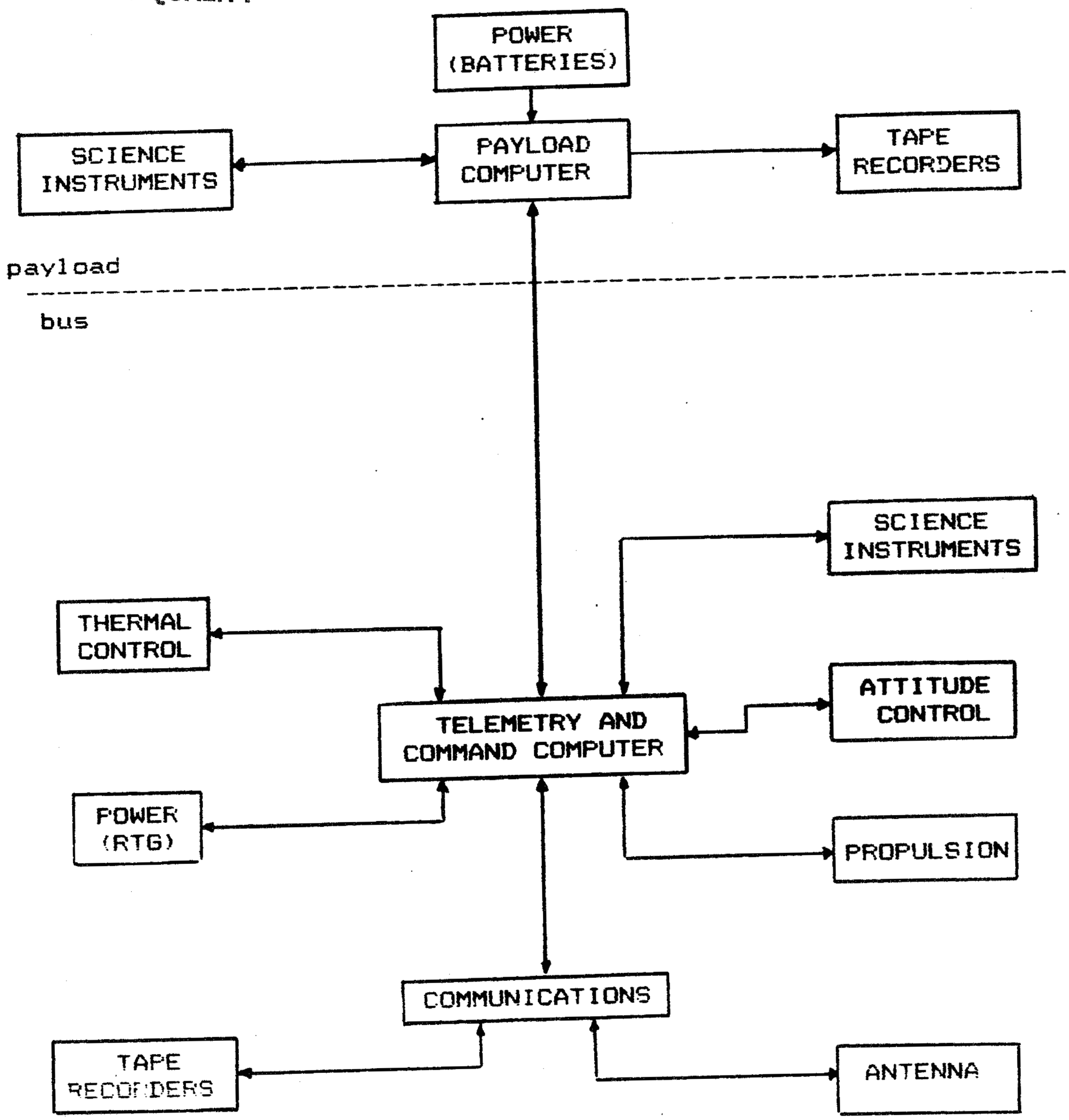

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SCIENCE INSTRUMENTATION (SCIN)

As complex as mission planning can be, it would be incomplete if science experiments were not conducted on the mission. An excellent design must take advantage of the primary mission (Mars Airplane Delivery) to accomplish as much as the time and budget will allow. For this reason, science instrumentation **(scin)** has been selected to go on the mission and designed to satisfy a pre-assigned set of requirements. Overall there are three purposes for the experiments which will contribute to our knowledge of the universe by taking advantage of the craft leaving the Earth-Moon system. These are to:

1) BETTER UNDERSTAND THE SOLAR SYSTEM
2) BETTER UNDERSTAND THE EARTH
3) BETTER UNDERSTAND THE APPEARANCE OF LIFE

In addition to any system requirements previously mentioned in the report, **scin** has its own set of sub-system requirements to meet. Once these are understood, a method for attacking and satisfying them can be derived. The first requirement is also the most important for mission completion, it is to autonomously verify the condition of the landing site for the descent vehicle. This guarantees that the pre-determined landing target has not been naturally destroyed before payload separation. The second requirement is the substance of the **scin** design, it is to complete six mission science objectives to augment aircraft science data. These are:

SOIL: Determine elemental and mineralogical surface character
DUST: Locate and map areas of high dust and volatile material content
GRAV: Define global gravitational field
TOPOG: Measure and map global topography
ATMOS: Explore atmospheric structure and circulation
MAG: Establish the nature of the global gravitational field

It has been determined that aircraft support will take precedence over global mapping. For this reason the instruments will operate mainly when the airplane is within their field of view and will verify airplane data. Global data collection will occur according to data storage capacity available (refer to **CCC**) and completeness of coverage needed. This need not be more than 10 complete orbits to satisfy global mapping requirements. The third **scin** requirenment is a direct result of the first two but essential for the overall spacecraft design and success of the mission, it is that **scin** must interact with all other pertinent sub-systems. This is shown schematically in Fig. 7.1.

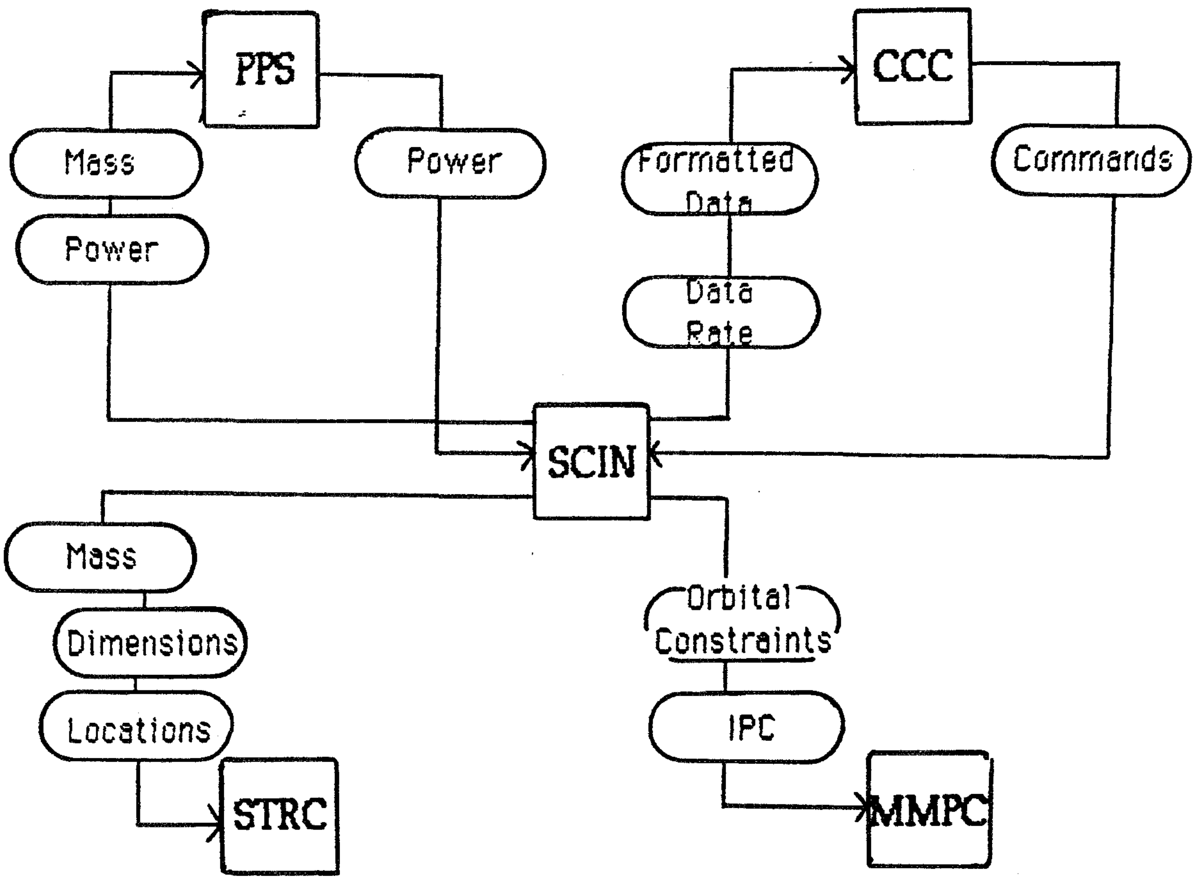

Fig 7.1

The <u>method of attacking</u> and satisfying **scin**'s requirements is relatively simple because it involves mostly research and very little calculation. The first move of the attack is to compile a large list of possible instruments for the mission. The sources of the list come from researching previously similiar missions. Next, scientists are interviewed to determine exactly how the instruments operate and what they accomplish. In this manner, it can be seen which instruments will contribute to the completion of a science objective. At this point a two step revision of the list is conducted to arrive at the final list. First all holes in requirement completion must be filled; this entails further research. Then any instruments that possess overlapping functions go through a comparison study of various categories in which the better instrument is retained. This is demonstrated in flow chart form in Fig. 7.2. The resulting instrumentation list must pass all sub-system interaction constraint considerations before it becomes final.

COMPARISON STUDY

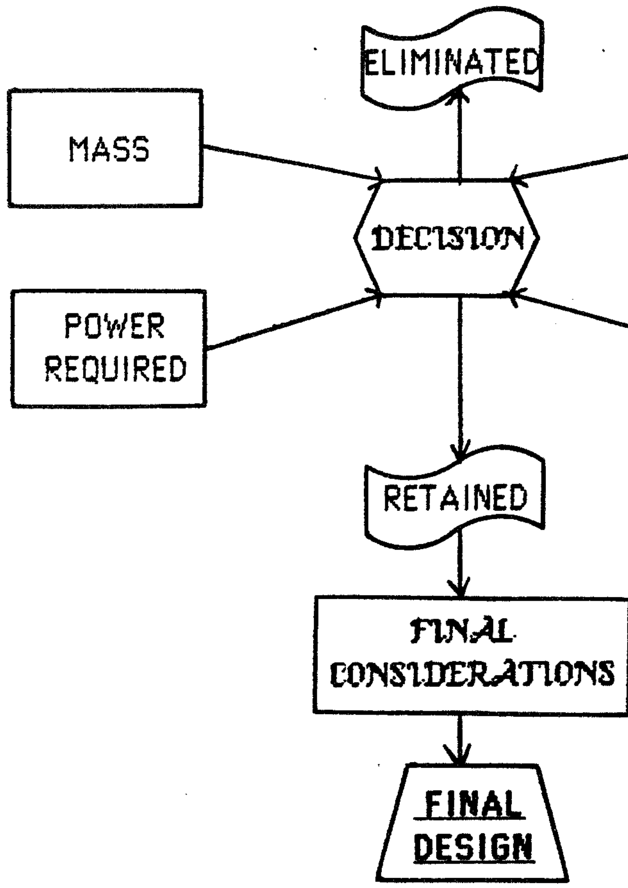

Fig. 7.2

The final instrumentation list is shown here. A detailed description of each instrument begins on page 7-4.

GAMMA-RAY SPECTROMETER
IR MAPPING SPECTROMETER
DUST DETECTOR AND COUNTER
FREQUENCY TRANSMITTER
RETARDING POTENTIAL ANALYZER
AUTO ALTITUDE EVALUATOR
MAGNETOMETER
IR THERMAL MAPPER
UNIVERSAL CAMERAS
RADAR ALTIMETER
MASS SPECTROMETER
PRESSURE MIRROR
ATMOSPHERIC SENSORS

Recalling the primary **scin** requirement of landing site verification, visual imaging will be used to certify that the present state of the site is suitable for a landing. The imaging wil be done by the universal cameras and the go-ahead command for payload separation will be done by artificial intelligence. The cameras will be uniformly inclined (thus UIUC) by the scanning platform and point at the target as the craft moves through its precessing orbit over the site (refer to **MMPS**). The visual

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footprint projected to the surface passes directly over the site every 3380 km/(2.88 km/sec) = 1173.61 sec. with a desired resolution of 40 km[15]. The scanning platform will be slewed by computer to produce proper images (focusing). Once conditions have been verifed, the payload will be separarted for atmospheric entry.

The satisfaction of scin's second requirement is broken up here into the six science objectives (experiments).

EXPERIMENT 1: SOIL-Determine elemental and mineralogical character of surface.

Gamma-Ray Spectrometer

The Gamma-Ray Spectrometer measures the gamma-ray intensity emitted from the surface. Because each element has its own set of characteristic gamma-ray lines, the identification and abundance of the surface elements can be determined. Volatile materials can also be detected in this fashion. The instrument will be mounted on a boom of at least 10 m[d] to minimize any spacecraft sources of radiation. Immediate gamma-ray flux is received from an 8' (1158882.7 km^2)[1] circle on the surface. The spectrometer has a field of view (FOV) of 120'[2] and must remain pointed at the surface while on the boom. Data is stored in two memory banks (one is used as a buffer) at a rate of 600 bps[3] each. While data is being recorded in one, the other bank is feeding data into **CCC** and erasing its memory. A radiator is required to cool the instrument, it also is located on the boom. The entire package draws 10 W[3] of power and weighs 13.5 kg[3]. The size of the sensor is estimated to be101.6 x 48.3 x 43.2 cm[21] with an electronics package of 22.4 x 24.8 x 21.2 cm[21]. Inheritance Class Percent (ICP) = 100% exact repeat. (refer to **MMPC**)

IR Mapping Spectrometer

The IR Mapping Spectrometer is used to locate surface areas of high water concentration. The instrument detects spectral reflectance of 1.38 um (the wavelength of water vapor infrared radiation) in order to locate much needed water sources. The sensor is extremely sensitive to light and is constrained to point directly at the surface without any obstructions or stray light. For this reason it is mounted on the scanning platform and will collect data at a rate of 3800 bps[4] in the day and 1100 bps[4] at night when there is less spectral reflectance. It can be programmed for a resolution of 1.3 km[4] within its FOV of 0.1' x 6.6[2]. The dimensions of the instrument are 71 x 28 x 20 cm[14], it weighs 21.7 kg[3] and draws 11.4 W[3] of power. ICP = 100% exact repeat.

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IR Thermal Mapper

The IR Thermal Mapper operates in the same fashion as the IR Mapping Spectrometer but will be used to detect and identify surface mineralogical formations. Rock-forming silicates, salts, and petroleum infrared brightness of 6-24 um will be detected to map these and any similiar formations. Volatile substances within this range can also be identified and mapped. The mapper will also be mounted on the scanning platform to fascilitate its pointing constraints of only a 1' x 1.5' FOV[2]. Data is collected at a rate of up to 4314 bps[5] with a surface resolution of 3 km[6]. The instrument performs a substantial amount of its own on-board processing with an analyzer to accomodate the down-link rate. The entire instrument (sensor plus analyzer) is 30.5 x 23.5 x 19.2 cm^3[7], weighing 10.7 kg[3] and drawing 13.7 W[3] of power. ICP = 100% exact repeat.

EXPERIMENT 2: DUST-Locate and map areas of high dust and volatile material content.

Dust Particle Detector and Counter (DPDC)

The DPDC is a simple instrument that has been used and proven on many previous missions such as Comet Rendezvous and Saturn Orbiter. Its sole purpose is to detect and count dust particles much in the same way a geiger counter detects radioactivity. The instrument must point toward the surface to obtain accurate readings and map dust concentrations but because its FOV is not available it will be mounted on the inside face of the orbiting bus. Data will be collected at a rate of 40 bps[18] with a resolution of 10 km[d]. The size of the instrument can vary greatly, the one designed for our mission has been estimated to be 30 x 20 x 20 cm[d], weighing 5 kg[19] and drawing 5 W[19] of power. ICP = 50% exact repeat; 50% minor modifications.

Gamma-Ray Spectrometer- see above

IR Thermal Mapper- see above

Pressure Mirror- see below

EXPERIMENT 3: GRAV-Define global gravitational field.

Ultrastably Accurate (USA) Frequency Transmitter

The USA Frequency Transmitter is a unique device that utilizes radio waves to measure gravitational pull. A signal transmitted from Earth at a precise frequency is received by the transmitter and sent back

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to Earth at the same frequency. The "reflected" signal is received and analyzed on Earth, and the magnitude of the Dopler shift on the signal is a direct function of the gravitational effects on the spacecraft. Special attention must be exercised on the transmission sequence so that no other bodies interfere with the signal. For this reason, the USA Frequency Transmitter will be mounted near the **CCC** down-link antenna for pointing accuracy. Any dangers in signal crossing or destructive interference are inherently corrected by the frequency chosen. The instrument weighs only 1.5 kg[3] and draws 3 W[3] into a cubicle 45 x 20 x 20[22] cm geometry. ICP = 100% exact repeat.

EXPERIMENT 4: TOPOG-Measure and map global topography.

Uniformly Inclined Universal Cameras (UIUC)

The UIUC imaging is the same used for the previously mentioned site verification. Two cameras used so that image frames are constantly overlapped and pieced together in a stereographic manner to form a complete view of the surface features. While the shutter of one camera opens, the other camera is midway through its scaning sequence. With the surface appearing to be moving under the spacecraft at a rate of 2.88 km/sec, a snapshot will be taken by both cameras every 25.91 sec[d]. This corresponds to the cameras FOV which combined is 3.1' x 1.51'[15] or 74.63 km x 36.36 km area from 1380 km. Because of the nature of recording image lines that make up a picture frame, the camera's data rate will be approximately 56,000 bps[d]. The entire system weighs 11.5 kg[d], draws 7.5 W[d] and takes up 50 x 25 x 25 cm[d] on the scanning platform. ICP = 20% exact repeat; 80% minor modifications.

RADAR ALTIMETER

The radar altimeter is another highly proven instrument used on past missions to obtain a relief map of the surface. It utilizes a 45 x 45 x 10 cm[8] box of electronics attached to a 1m[8] x 22[d] cm dish antenna which remains pointed toward the surface. The signal sent down to the surface has a footprint of 115882.7 km^2 circle ($\pm$8')^2, and is reflected and recorded on board at a rate of 576 bps[3] (average). It can be programmed to collect data at a rate of up to 1440 bps[3] when greater surface detail is desired. The entire instrument weighs 17 kg[8] and draws 28 W[8] of power. ICP = 90% exact repeat; 10% minor modifications.

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EXPERIMENT 5: ATMOS-Explore atmospheric structure and circulation.

Retarding Potential Analyzer

The RPA will be mounted on the payload vehicle with its sensor right on the face of the aeroshell. It measures the current or electrically charged particle density from 10-10^6 part/cm^3 in the atmosphere in order to identify ions throughout atmospheric descent. The depth of solar wind electricity in the atmosphere is a direct function of magnetic field strength. It is activated shortly after separation and receives 2.8 W[20] from a local battery power source while accumulating data as a rate of 500 bps[1] to be recorded on a local tape recorder (refer to CCC). The dimensions of the instrument are 28 x 14.5 x 14.5 cm[16] and it weighs 2.7 kg[20]. FOV need not be discussed for any of the atmospheric instruments which only make instantaneous measurements. ICP = 90% exact repeat; 10% minor modificatons.

Mass Spectrometer

Like the RPA, the Mass Spectrometer goes down to the surface on the payload vehicle with its sensor mounted on the aeroshell. The Mass Spectrometer ionizes neutral particles in the atmosphere with an electron beam so that their momentun (and in turn mass) can be measured. Ions with 1 to 50 AMU[17] can be measured qualitatively and quantitatively from 250 km-100 km. The instrument draws 13 W[20] from the local battery power source and collects data at 500 bps[18] to be recorded on the local tape recorder. Dimensions of the Spectrometer are 50 x 35 x 35 cm[16], weighing 4.3 kg[20]. ICP = 100% exact repeat.

Pressure Modulator IR Radiometer (Pressure Mirror)

The Pressure Mirror is a multi-dimensional instrument that performs simultaneous measurements to plot 3-dimensional profiles as a function of atmospheric pressure. Thermal structure is mapped from 0 km -80 km, atmospheric dust is profiled, verticle distribution of atmospheric water is mapped from 0 km-35 km, and atmospheric pressure is profiled, all simultaneously with a resolution of 5 km (verticle)[10]. Data is collected at a rate of 155 bps[3] within a FOV of 0.9' x 1.6'[2]. Automatic calibration is inherent by taking a deep space zero level reference reading. The instrument is mounted on the orbitng bus facing the planet, weighing 25.7 kg[3], and drawing 27.1 W[3] of power. The Pressure Mirror has dimensions of 75 x 40 x 30 cm[23]. ICP = 100% exact repeat.

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Automatic Altitiude Evaluator (AAE)

The AAE is a simple altimeter placed on the payload vehicle for the purpose of triggering the local power source which activates atmospheric instrumentation. The previously mentioned RPA and the Mass Spectrometer will be activated when the AAE hits 250 km and de-activated at 100 km. At this altitude the simpler sensors (ST. PATS, see below) which are designed for lower altitude measurements and profiles are activated and collect data all the way to the surface. The AAE is a 22.9 x 7.6 cm[24] cylinder that weighs 9 kg[24] and draws approximately 5 W[24] of power. ICP = 100% block buy.

Space Transport Pressure, Acceleration, and Temperature Sensors(ST. PATS)

ST. PATS are common laboratory instruments placed on the payload vehicle to take lower atmosphere measurements of dynamic pressure, acceleration, and temperture. Data will be collected at 20 bps[d] and recorded on the local tape recorder for profiling. The entire system will draw less than 5 W[d] from the local battery source, weighing less than 5 kg[d], and is estimated to 10 x 10 x 10 cm[d]. ICP = 100% block buy.

EXPERIMENT 6: MAG-Establish the nature of the global gravitational field

Magnetometer

The magnetometer is an instrument that measures the magnetic flux passing through its sensor. It must be mounted on a boom of at least 7 m[11] to minimize the spacecraft magnetic field. An "inboard" sensor located at 3.5 m[12] will be used to measure the spacecraft magnetic field and calibrate the (main) "outboard" sensor. Both sensors are 10 x 7.5 x 7.5 cm[13], weighing 1.25 kg[13], and are connected to an electronics package (20 x 15.8 x 10 cm on the bus each[13]) weighing 1.94 kg[13] (sensor specifications are updated and inconsistent with STRC configuration). The entire package draws 2.45 W[13]. The sensor is a fluxgate that has no pointing constraints, passing through the magnetic field allows it to collect data anywhere from 80 -320 bps[13](programable). The fluxgate simulates a 180' rotation to eliminate zero level drifts. IPC = 100% exact repeat.

RPA-see above

The third **scin** requirement of sub-system interaction demonstrated earlier is inherently satisfied in the completion of the spacecraft design.

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As in all designs, a set of problem areas are developed which require further investigation. For **scin**, 3 such areas have been conceived. First, the problem of global mapping must be resolved because the precise amount of time required to accumulate sufficient global data has not been determined. This requires a complete study of surface footprints for each orbit and the FOV of the instruments. Then the computer can be programmed to activate global mapping instruments at specific times until sufficient data is acquired. Second, the UIUC data rate and imaging techniques require further research so that the techniques of utilizing two cameras can be optimized and a more specific data rate can be quoted. Third, instrumentation mounted on the orbiting but not fixed to the scanning platform must be accomodated for in the future design. Although pointing requirements for such instruments have been satisfied, using a larger scanning platform and placing all of the instruments on it will provide uniformity in surface viewing.

REFERENCES

1) **Mars Observer**, "Investigation Description and Science Requirements Document", 1987.

2) **Viking**, Viking Mission to Mars, 1974.

3) **Mariner Mark II**, "AIAA-84-0214, Mariner Mark II Program", 1984.

4) Ralph Parrish, JPL Instrument Engineer, [redacted]

5) Len Tyler, **MO** Principle Investigator, [redacted]

6) Larry Majorana, JPL Instrument Engineer, [redacted]

7) John Nitz, USAF Flight Engineer, [redacted]

FOOTNOTES

1) ref 1, p 2-14
2) ref 1, p 4-6
3) ref 1, p 4-5
4) ref 1, p 4-47
5) ref 1, p 4-40
6) ref 1, p 5-13
7) ref 1, p 2-60
8) ref 1, p 2-49
9) ref 1, p 4-11
10) ref 1, p 2-34
11) ref 1, p 4-2
12) ref 1, p 2-18
13) ref 1, p 2-19
14) ref 2, p 46
15) ref 2, p 43
16) ref 2, p 50
17) ref 2, p 51
18) ref 3, p 18
19) ref 3, p 10
20) ref 3, p 11
21) ref 4
22) ref 5
23) ref 6
24) ref 7

d) dimensions are designed and unique to our spacecraft

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PROJECT ACRONYM

by
Group #5

AAE 241
Spring 1988

Peter Rachesky..............Mission Management
Sanjeev Dhand...............Power and Propulsion
Russell Wenzel..............Attitude and Articulation
Jeff Bradshaw...............Structures
Sisir Kudva.................Command and Communication
Toby Martin.................Science Instrumentation
Larry Kim...................Aerobrake

Peter Rachesky [redacted]
Sanjeev Dhand [redacted]
Toby B. Martin [redacted]
Russell S. Wenzel [redacted]
Jeff Bradshaw [redacted]
Larry Kim [redacted]
Sisir A. Kudva [redacted]

RACHESKY
GROUP 5

The trajectory chosen from Earth to Mars was a Hohmann transfer ellipse. The ideal ellipse has a true anomoly of 180 degrees, the major axis starting at Earth and ending at Mars, running along the line that joins the conjunction and opposition, the same line that divides the Eastern and Western quadrature. This would be the ideal ellipse because the energy used would theoretically be lowest, allowing more mass for instruments and payload instead of fuel. There are other factors to consider, however: Earth/Mars communication distance, flight time, the season of the surface of interest, the relative position of other planets, hemishperical launch direction if leaving from earth, etc. Because there are other considerations, the trajectory is rarely ideal. However, the energy deviation just short or just long of the ideal transfer is small and weight can be given to other considerations.

CONSIDERATIONS	USES	PROBLEM
FLYBY OF PLANET, HERE VENUS	FLYBYS CAN DRASTICALLY LOWER ENERGY NEEDED FOR LAUNCH	IN THIS CASE, THE ENERGY NEEDED FOR A FLYBY IS, IN ALL LAUNCH WINDOWS, GREATER THAN FOR A STRAIGHT SHOT TO MARS
GOOD RADIO COMMUNICATIONS	GOOD DATA RECEPTION AND COMMAND SIGNALS FROM EARTH TO CRAFT	LONG DISTANCES MAY DISTORT THE SIGNAL AND INCREASE TRAVEL TIME, AND SUN OR OTHER BODY MAY BLOCK SIGNAL IF ORIENTATION IS POOR
DELTA-V	THE LOWER THE VALUE, THE LESS FUEL IS NEEDED	LOW DELTA-V IS POSSIBLE FOR ONLY LIMITED NUMBER OF LAUNCH WINDOWS AND FLIGHT TIMES

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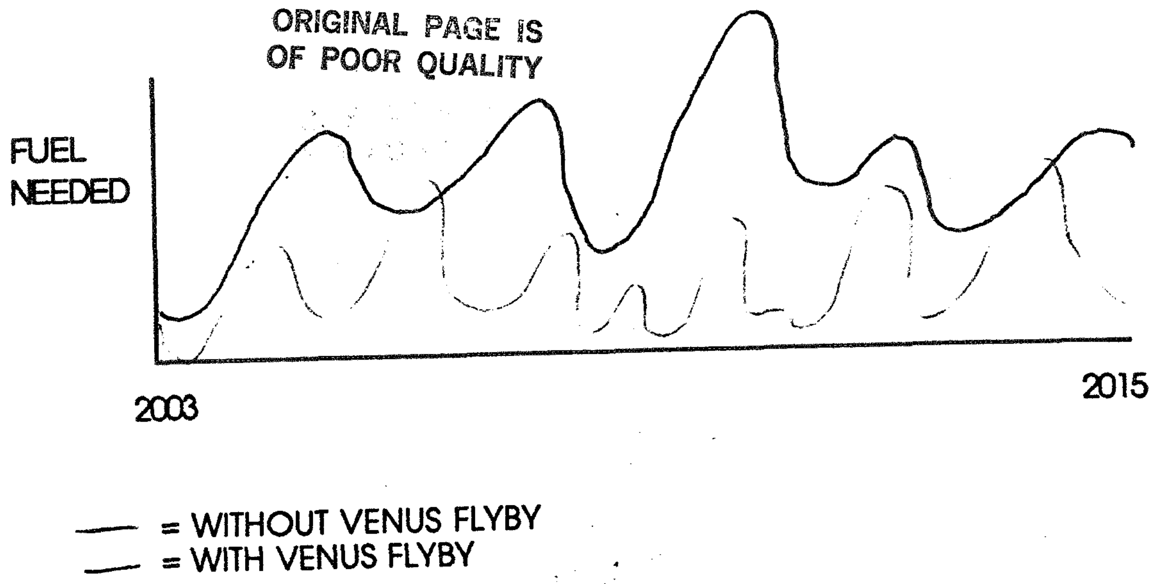

The use of a flyby also creates the added problem of putting the spacecraft closer to the sun, creating the problem of extra radiation and too much heat. As it turns out, the trajectory that has the lowest ΔV is excellent for radio communications, does not involve a flyby, and has, of the two options (short or long) a flight time short of the ideal.

The trajectory chosen is the best choice considering any one of those criteria mentioned in the block, and for that reason it is superior for the reasons combined. On the more technically oriented side:

TRAJECTORY DESCRIPTION

Spacecraft departs from an Earth parking orbit 7 June 2003, 450 km above sea level with an initial ΔV of 3.563 km/s (kps), making a velocity of 33.363 heliocentric kps.

DEPARTURE DATA

Ecliptical departure velocity w.r.t. Earth	3.563 kps
Declination (degrees)	-7.764

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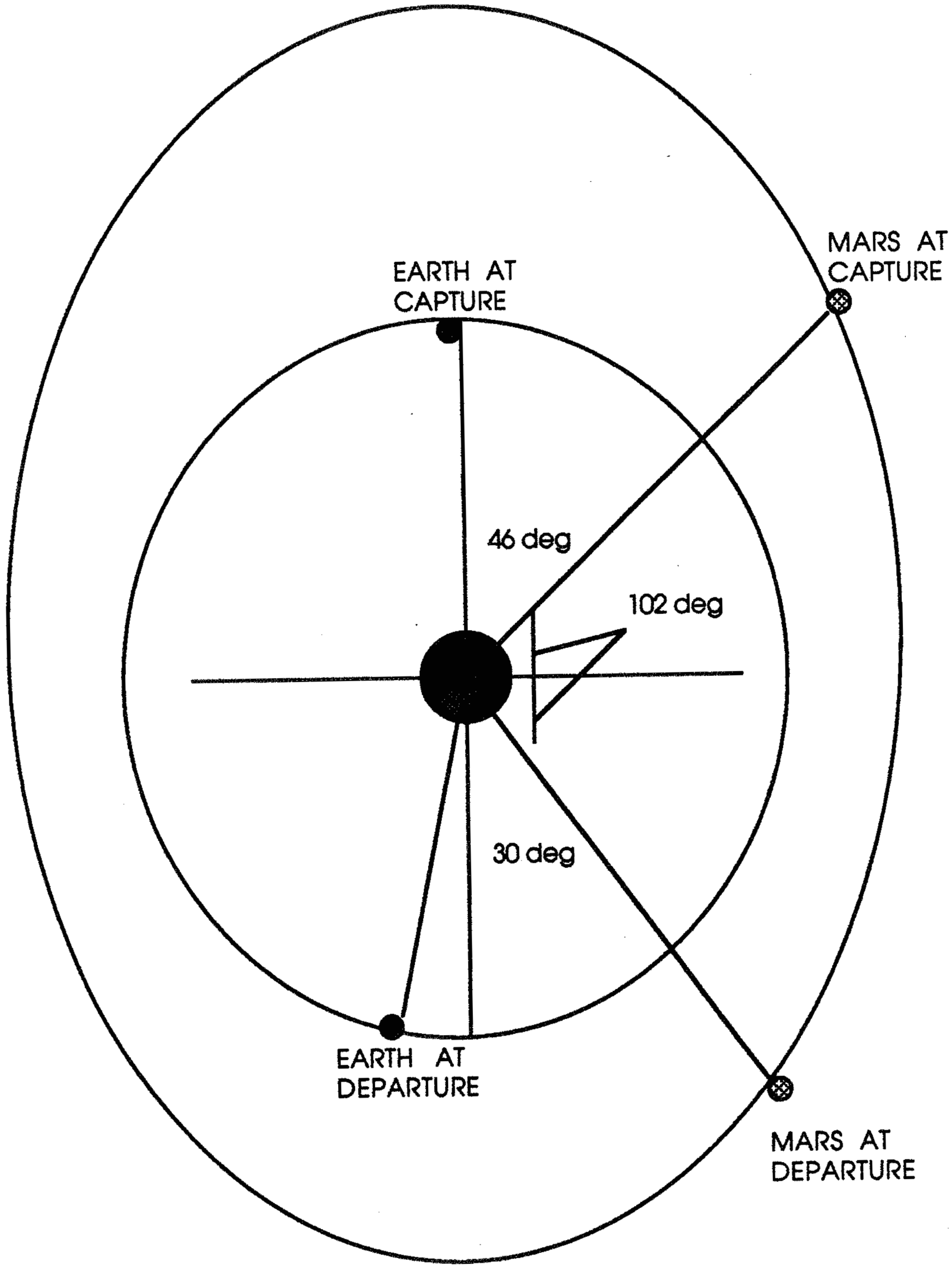

ECCENTRICITY OF MARTIAN ORBIT= .0933
ECCENTRICITY OF EARTH ORBIT ALMOST NEGLIGIBLE

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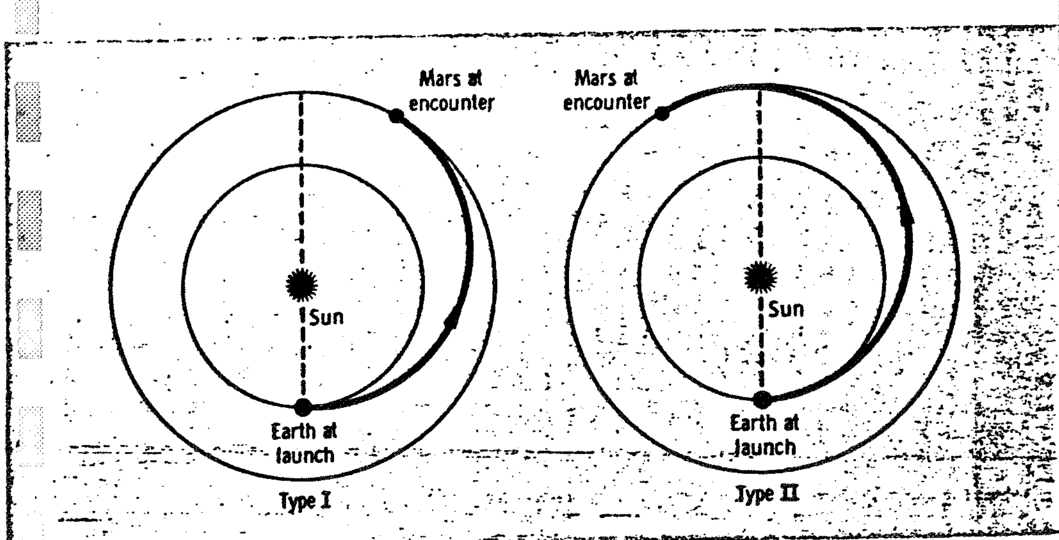
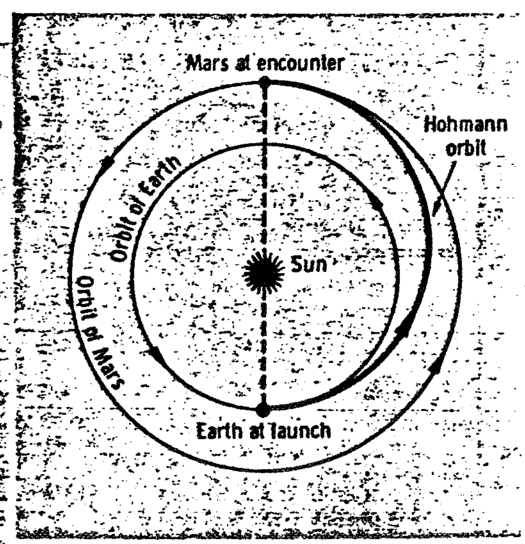
Right Ascension (deg)	348.276
Phase angle (VHP and Sun-Earth Vector included angle $0 \leq \Phi \leq 180$) (deg)	90.020

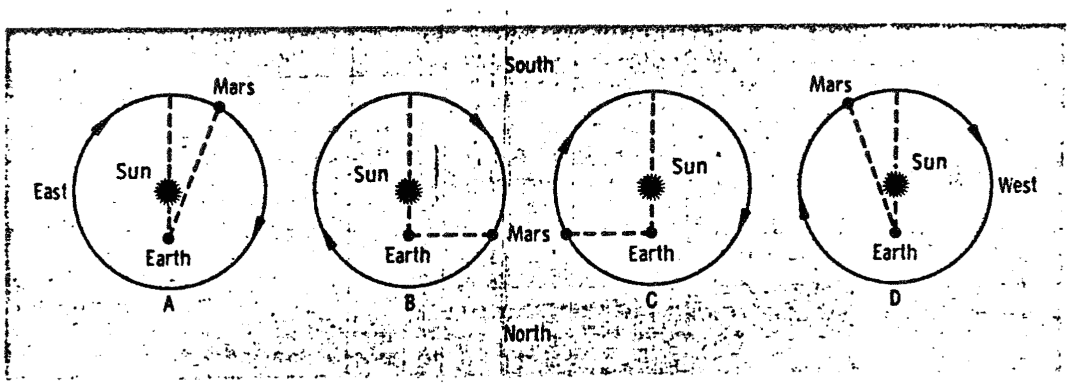
Craft arrives at Mars 198.4 days and 572 million km later on 23 December 2003, 620 km from Martian surface at periapse (44,260 km when at apoapse on first aerobrake ellipse) at heliocentric velocity of 32.173 kps.

ENCOUNTER DATA

Ecliptical encounter velocity w.r.t Mars	8.073 kps
Declination (degrees)	7.949
Right Ascension (deg)	59.187
Phase angle (VHP and Sun-Mars Vector included angle $0 \leq \Phi \leq 180$) (deg)	82.586
Sun-Earth-Mars angle (deg)	93.013
Radio Communications Distance to Earth	154.507×10^6 km or 8-9 minutes lag time







(Shown above, the Sun-Earth-Mars angle)

<u>RELATIVE DEPARTURE COORDINATES</u> (IN ASTRONOMICAL UNITS)

(1 AU = 149.6 million km = mean orbital radius of Earth)

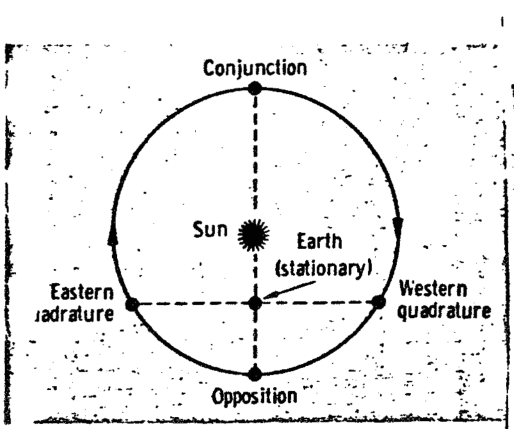
DEPARTURE-

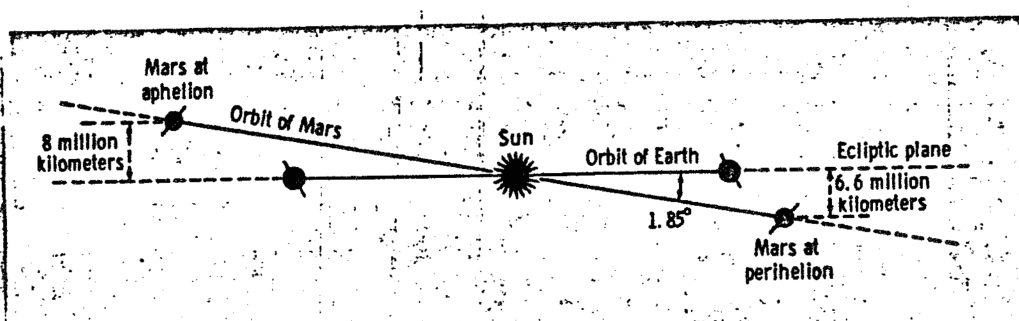
X= -.2510 Y= -.9384 Z= .0001 Distance from Sun = 1.1049

ENCOUNTER-

X= 1.0309 Y= 1.0384 Z= -.0031 Distance from Sun = 1.4633

The ellipse of this trajectory makes a near perfect insertion orthogonal to the radial direction of Mars as shown by the phase angle of 82.586, off by (90 - 82.586)= 7.414 degrees. Although the trajectory path is basically in the ecliptical plane, there is a change in the Z coordinate of 478,720 km. This is due to the fact that the orbital plane of Mars deviates from the ecliptical plane an average of 1.85 degrees. This Z change is in itself quite small, as the Martian orbital plane varies from the ecliptical by as much as 8 million km at apehelion.





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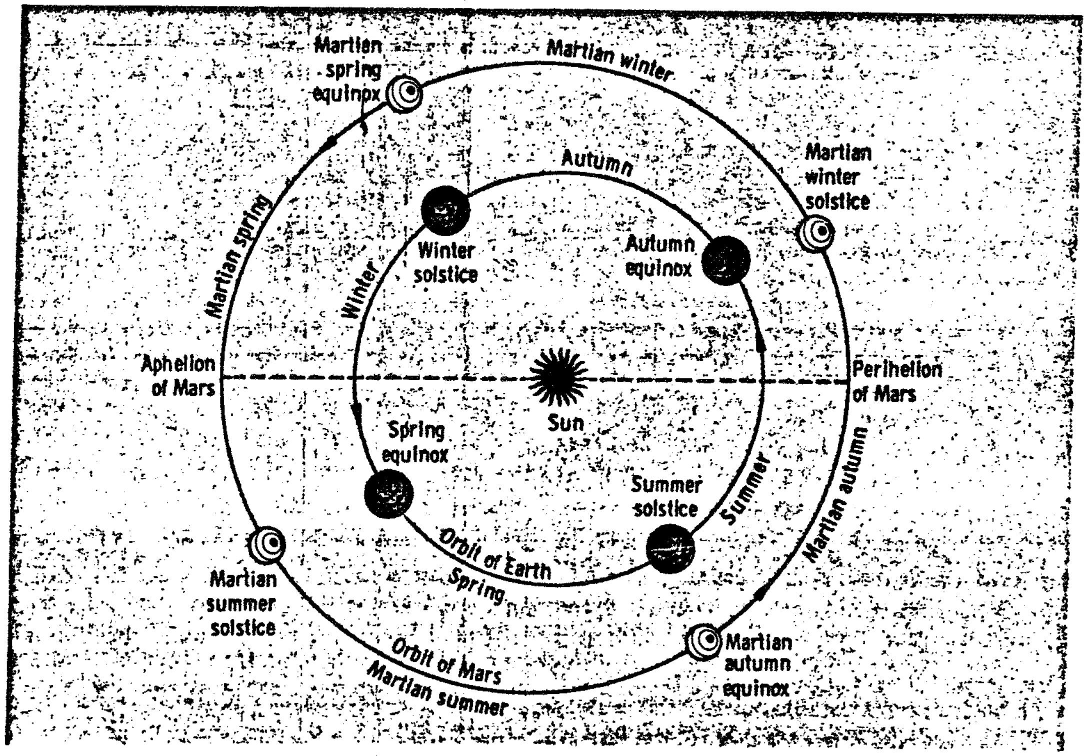
A measure of the energy (hence fuel) required while in orbit has been mentioned above, the ΔV, or instantaneously approximated change in velocity. Obviously the lower this value, the less fuel need be carried. This is also one of the main reasons the aerobrake will be used, as discussed later. An overall value of ΔV with breakdowns is shown below:



(shown above: various surface seasons mentioned as criteria for launch)

MISSION ΔV REQUIRED

Earth Orbit Departure	3.563 kps
Mars Orbit Capture	.816 kps
Aerobraking Manuever	.168
Manuevers to Circularize to synchronous orbit	1.66
Maneuvers to Equatorial Synch	.632
Total=	6.839

PHASE 1: 1 JANUARY 2003–
FIRST SHUTTLE TO EARTH STATION– FULL SHUTTLE PAYLOAD

PHASE 2: 24 JANUARY 2003–
SECOND SHUTTLE TO EARTH STATION- HALF SHUTTLE PAYLOAD

PHASE 3: 7 JUNE 2003--
OTV MOVES SPACECRAFT AWAY FROM STATION– SPACECRAFT MAKES FIRST IGNITION OF 3.563 KPS ON HOHMANN TRANSFER TO MARS

PHASE 4: 11 JUNE 2003–
FINE TUNING ADJUSTMENT OF TRAJECTORY/ 180 DEGREE ROTATIONAL MANUEVER

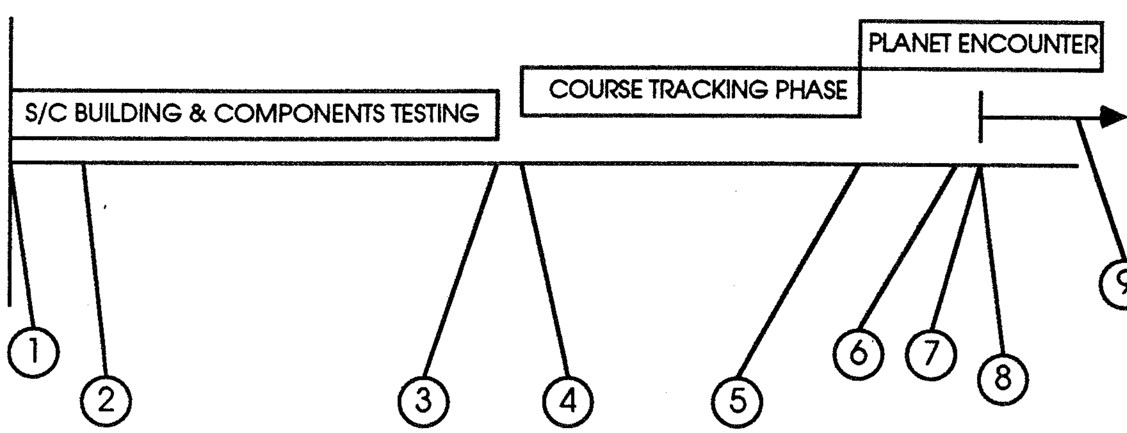
PHASE 5: 23 DECEMBER 2003--
MARS ENCOUNTER; IGNITION OF .816 KPS TO ELLIPSE OF ECCENTRICITY =.833, PERIAPSE OF 4000 km, SPEED OF 8.073 KPS; BEGIN AEROBRAKING AND PREPARE FOR SCIENCE DATA COLLECTION

PHASE 6: 16 FEBRUARY 2004–
CIRCULARIZATION OF ORBIT, CARGO DESCENT, PAYLOAD/BUS SEPARATION

PHASE 7: 19 FEBRUARY 2004–
TRANSFER TO SYNCHRONOUS ORBIT

PHASE 8: 20 FEBRUARY 2004–
CARGO DOWN; AWAIT ASSEMBLY TO TRACK PLANE; SWITCH TO EQUATORIAL SYNCHRONOUS ORBIT AND CONTINUE SCIENCE DATA COLLECTION; MONITOR TO END OF MISSION

PHASE 9: 1 APRIL 2004 END OF MISSION



MISSION DESCRIPTION

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TRANSPORT TO STATION

The spacecraft is estimated to be able to fit into one and one half shuttle cargo bays. Therefore, it will take two lauches of the shuttle before the whole of the craft is together. The spacecraft will be designed in such a way as to facilitate ease of assembly with the materials aboard the space station. Estimated assembly completion time including testing of the more sensitive items is expected to take three months. A pad of two months is allowed for unforseen problems, as the missing of the launch window could paralyze the spacecraft at the station until the next available launch window.

LAUNCH FROM EARTH

An orbital transfer vehicle standard to the space station will be used, as this will probably be standing operational procedure for protection of the station against malfunctioning equipment. The OTV will push the spacecraft to a higher orbit from which the spacecraft will commence its first burn of 3.563 kps with the direction of the Earth at an angle orthogonal to the Sun-Earth vector plus .02 degrees. This is the phase angle at departure, with a heliocentric velocity of 33.363 kps, which includes the 29.8 kps at which Earth travels around the Sun. The trajectory is basically in the ecliptic plane with minor deviations as noted above accounting for the deviation of the Martian orbital plane to the ecliptic by 1.85 degrees. This also takes into account the tendency at the time of launch from an equatorially orbiting station the effect of 'throwing' the spacecraft above the ecliptic plane because of the direction of rotation of the Earth.

ENROUTE TRANSIT

The spacecraft leaves the gravitational sphere of influence of the

Earth and becomes a satellite of the Sun. After approximately four days post launch, the spacecraft's orbit is adjusted or 'fine tuned', and the craft is rotated 180 degrees in what is called the midcourse manuever in preparation for Martian capture. No other propulsive burn should be required until the capture, but should one be necessary, the attitude and control systems have more than enough in reserve for even a few adjustments along the way. Until the craft reaches Mars, we track and wait. Telemetry is sent back to Earth periodically to verify headings and to report on the health of the craft.

MARS CONTACT AND AEROBRAKING

The craft contacts Mars on 23 December 2003 and adjusts to fine tune from its aiming point just off the surface of the planet. A retro burn of .816 kps is expended and now the one stage liquid hydrogen/oxygen propulsion stage is jettisoned exposing the monomethylhydrazine/nitrotetraoxide stage. Aerobraking commences here, on an ellipse of e=.833, periapse of 4000 km (620 km from the surface) and later reaches its first apoapse of approximately 44,000 km. The aerobraking procedure requires about 6.8 days according to the aerobrake subgroup to complete its energy dissipation and orbit circularization. A higher circularized orbit is desired to reach synchronous orbit, so another Hohmann transfer is utilized (1.66 kps) to bring the entire configuration to an orbit of approximately 6000 km from the center of Mars.

CARGO/BUS SEPARATION

At this time, the cargo breaks from the instrument bus and each continues in its path, much like a train with a broken couple in a frictionless environment. The bus monitors the surface and sends data back to the Earth, forming the basis for the decision as to when the cargo is dropped. Once the cargo is safely on the ground, the instrument bus changes from an ecliptical orbit to a synchronous ecliptical, then an equatorial synchronous orbit and prepares to monitor the aircraft as it takes its required science readings. This is to be the terminal orbit, and it will be synchronized after all data is collected to make it easy to shoot a directional azimuth for radio communications. The satellite remains in orbit to monitor the movement of the aircraft.

PROBLEM AREAS

There are, inherent in all planned missions, problem areas. But foreseen problems can be planned for. It's the lack of foresight and troubleshooting that can lead to disaster.

Upon encounter with Mars, it is possible (though the chance is very remote) that the craft could hold a suicide heading toward the inner moon of Phobos. Phobos is a Martian satellite 16 km in diameter and 6100 km from the surface of the planet inclined at 1.13 degrees to the equator. Its period is very small, revolving around the planet once every seven and one half hours. It is the only satellite that revolves around a planet faster than the planet rotates in our solar system. Its orbit is also so close that a Martian observer standing north of 70 degrees latitude would not be able to see Phobos. For these reasons, the little moon is very dangerous considering the vast numbers of orbital manuevers and orbital trajectories around Mars. Should a collision course be detected, a change of trajectory with much recalculation back on Earth should be in order.

Also is the possibility, however remote considering fifty years of reliability, that the Martian capture engine won't reignite. An attempt was made to make the possibility of a return using attitude and control and the hyperbolic speed already present with the craft, but it was found that a "Mars flyby" return to Earth was not possible. Should the engines fail, the mission is scrapped. Fifty reliable years is nothing to take lightly, however. Engine failures of this nature have yet to be recorded in recent history.

COST ANALYSIS

There is a function relating mass and cost directly. The entire project is broken down into two categories for costing purposes:

1) Functional support category- that cost which includes such project divisions as system support and ground equipment, data development, management, flight operations, and data analysis. These divisions deal more with people and the 'big picture'.

2) Hardware related category- that cost which can be directly related to the maintenance, testing, development, or design of an actual single piece of hardware.

Each subsystem and major end item component is analyzed separately for the labor hours it requires. Since labor hours are the specific dependent variable, they are used as a decoupler in the natural dependency of inflation over time. Simply change the hourly wage to what is current, and the functional algorithms still hold. By and large, each specific cost for each specific item is not very accurate. But when taken as a group sum, a very good estimate of cost can be obtained, provided there are no major delays and the schedule that is set is reasonable. The model is only as good as the data it's based on.

The model also separates each component into two categories: DLH or direct labor hours is the development and design of the component, and RLH or recurring labor hours is the fabrication and testing of each component. The reason for the separation is that now the model takes into account the savings incurred when an item can be purchased straight out instead of developed from scratch. The savings are as follows:

DLH SAVINGS	METHOD OF ATTACK
100%	BLOCK BUY
80%	SUBSYSTEM COPY
25%	MINOR SUBSYSTEM MODIFICATION
5%	MAJOR SUBSYSTEM MODIFICATION
0%	FRESHLY DESIGNED SYSTEM

m in kg

DLH, RLH in kilohours

DLH savings does not take into effect the purchase price however.

ED= encounter duration

MD= mission duration

HARDWARE

SINGLE STAGE PROPULSION SUBSYSTEM

propulsion m=12,950

DLH= 56.1878 $(3m)^{.4166}$ =4588 100%-cost=50%=2294

RLH= $(3m)^{.9011}$ =13662

structure m=1240

DLH=1.626 $(4m)^{.9046}$ =3581

RLH= 1.399 $(4m)^{.7445}$ =789

INSTRUMENT BUS

propulsion m=20,292

DLH= 56.1878 $(3m)^{.4166}$ =5531 50%=2766

RLH= $(3m)^{.9011}$ =20477

structure m=356

DLH=1.626 $(4m)^{.9046}$ =1158

RLH= 1.399 $(4m)^{.7445}$ =312

data handling and communications m=104

DLH= 4.471 $(4m)^{1.1306}$ =4088

RLH= 1.626$(4m)^{1.1885}$ =2108

computer control m=25

DLH= e^{4.2605 + .02414(4)(m)}=792

RLH= e^{2.8679 + .02726(4)(m)}=267

attitude and control m=190

DLH= 21.328$(4m)^{.7230}$ =2740

RLH= 1.932(4m)=1468

antenna m=104

DLH=6.093$(4m)^{1.1348}$=5715

RLH= 3.339(4m)=1389

science breakdown

altimeter m=30

DLH= 11.409$(4m)^{.9579}$ =1119 50%=560

RLH= 1.2227$(4m)^{1.2376}$ =456

remote sensing m=59

DLH= 25.948$(4m)^{.5990}$ =685

RLH= .790$(4m)^{.8393}$ =77

particle & field m=6

DLH=25.948$(4m)^{.7215}$ =257

RLH= .790$(4m)^{1.3976}$ =67

solar power m=48

DLH= e^{3.9633+ .00911(4)(m)}=302

RLH= e^{2.5183+ .02104(4)(m)}=705

SHIELD

aerodeceleration module m=600

DLH= 3.481$(4m)^{.8416}$ =2435

RLH= 4.662$(4m)^{.5}$ =228

CARGO BAY

structure m=3530

DLH= 1.626$(4m)^{.9046}$ =9227 100%-cost=50%=4614

RLH= 1.399$(4m)^{.7445}$ =1719

SUPPORT

system support and grnd equip

DLH=.36172(ΣDLHHardware)$^{.9815}$=8351

Flight Operations

DLH=(ΣDLHHardware)$^{.6}$*.00804(10.7MD+27ED)=1166

Lauch + 1 mos operations and grnd software

DLH= .09808(Σ DLH Hardware)=2736

Science data development

DLH= 27.836(science mass)$^{.3389}$=115

Data analysis

DLH=.425(Flight oper)=495

Program Management

DLH= .10097(Σ DLH all categories)$^{.9670}$=3180

Sum of all RLH=43724 Sum of all DLH=48030

Sum of all labor kilohours = 91,754

Conversion $10.45/hr

Conversion total labor to total cost =3.3

Total cost for four spaceships, 3 fueled, saving 50% on some DLH totals for shelf purchases= $3,164,142,000.00

RFP REQUIREMENTS AND COMPLIANCE

The requirements as set forth by the RFP Spring 1988, as can be seen, have been met. The performance has been kept at the lowest possible energy leaving science and communication much room. Off the shelf hardware was used where possible, however in such a new endeavor much of the equipment had to be designed top to bottom. Most materials are available for use now, much less 1998. It is not limited to an aircraft payload, and there is nothing precluding this vehicle from visiting another planet even, with only minor modifications for radiation and heat (Venus is the next logical planet). An OTV retrieves the spacecraft, satisfying the remote retreival criteria. Three systems have been assumed to be flight ready, while the no fuel was calculated into the cost of the fourth craft to remain on the ground. Our design stresses simplicity, using mostly proven, reliable equipment available already. As for artificial intelligence, that criteria must be answered by command/communications.

The flyby trade study, I must admit, was not much of a trade study. The were virtually no advantages whatsoever to the flyby, as pointed out earlier, so sensitivity criterion were never given a chance to come into play. There really wasn't any. Orbits around Mars were chosen by AACS, as his role was the manipulation of the instrument bus. The particular equatorial orbit chosen as final was a result of much debate weighing the primary priority of monitoring the aircraft with the secondary priority of mapping the poles of Mars. As can be seen by the final choice made between science and aacs, the aircraft won at the expense of a part of a science requirment.

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Project ACRONYM
Power and Propulsion Subsystem (PPS)

By
Sanjeev Dhand

Abstract

The Power and Propulsion Subsystem (PPS) for Project ACRONYM was designed with a heavy emphasis on reliability and availability of components. For these reasons, an all-chemical propulsion system was chosen. This system is divided into two parts. The first is a liquid hydrogen/liquid oxygen, high thrust phase that is in operation during Earth departure and Mars capture. The second phase uses MMH and NTO to provide orbit maneuvering and attitude control. The power system utilizes rigid solar arrays and Nickel-Cadmium batteries to supply the relatively low amount of power required by the craft. The batteries are sized for a geosynchronous orbit about Mars. Throughout the design, reliability and availability are strongly stressed.

Introduction

The Power and Propulsion Subsystem (PPS) for Project ACRONYM is designed to meet the requirements of the subsystem fully. These requirements include sending telemetry to and accepting commands from the Command and Communication Control subsystem. PPS must also be self-powered. It must sense such inputs as temperature and loads on its components, and it must control outputs like power relays and valve actuations. Just as all other subsystems, PPS must minimize cost and use only the technology available up to the year 1998. Since the instrument bus will be orbiting Mars for at least four years, all components selected must last at least that long, but nothing should prohibit them from functioning longer than that. The components for this subsytem include a power source, engines, thrusters, tanks, plumbing and valves.

A clear method of attack was chosen for this mission. First, an estimate of power and propulsion requirements was made using data from previous missions. Next, research and trade studies were performed to determine the most feasible sources of power and propulsion for our mission. Then, data was gathered from other subsystems to determine the required power and the delta-v's needed for the mission. Finally, components were selected and/or designed to meet these requirements. This is our current position in the mission to Mars. Next, the Power and Propulsion Subsystem will be tested to determine any flaws that may exist in the design, and finally, the design will be implemented.

Propulsion Subsystem

The propulsion subsystem for the Awesome Craft Really Overshadowing Next Year's Mission (ACRONYM) was designed with emphasis on reliability, availability of components, and cost efficiency. Chemical propulsion offered all three characteristics if used in our transport mission to Mars. With over thirty years of space flight experience, chemical is clearly the most reliable source of propulsion for the ACRONYM. Also, since almost every spacecraft since the development of space flight has used chemical means of propulsion, we will be able to use readily-available, off-the-shelf hardware. This will also minimize the cost for the ACRONYM's propulsion system. Although both electric and chemical means of propulsion were considered for use on the Earth-Mars Transport Vehicle (EMTV) and the instrument bus, chemical was chosen for both stages due primarily to its reliability and its feasibility for this particular mission. If electric-ion (low-thrust) propulsion was used, a very high amount of power would be required (approximately 300 KW). This would necessitate the use of a nuclear power source, or RTG's. I decided that since such a power source would not be available to us, an all-chemical propulsion system was more feasible.

The system is divided into two phases. The first phase is used to escape from Earth orbit and to capture Mars orbit. The second stage is used for orbit maneuvering and attitude control as well as circularizing the orbit at Mars following the aerobraking sequence. The first phase, requiring higher delta-v's, uses liquid hydrogen and liquid oxygen, providing a higher specific impulse (460 sec.). This requires us to make the assumption that the space station is capable of storing these liquid fuels. The second phase uses Monomethyl Hydrazine (MMH) and Nitrogen Tetroxide (NTO). These fuels were chosen for their availability and reliability after being used on so many flights of the space shuttle. Another advantage of this combination is that it is hypergolic.

The sizing of the propulsion system was performed beginning with the mass of the satellite bus at the end of the mission. This mass includes the bus structure, science instruments, communication instruments, and the empty tanks and thrusters. This mass was given as 1200 kg. Using the rocket equation:

$$\Delta V = g_0 I_{sp} \ln \frac{m_i}{m_f}$$

with I_{sp} = 340 sec. and a cumulative delta-v of 3.0 km/s including AACS requirements, the mass of the propellant was calculated as 2020 kg. By using a tankage factor of .06, a tankage mass of 120 kg is obtained. The instrument bus employs one large thruster for orbit maneuvers and several smaller thrusters for attitude control. Each thruster has two valves at the inlet to its combustion chamber, allowing us to operate each thruster individually. As shown in Fig. 1, the second phase propulsion system is a pressure-fed, MMH/NTO combination. The helium tanks prevent evaporation of the liquid propellant and oxidizer and keep the fuel tanks pressurized. This bipropellant requires an oxidizer-to-fuel ratio of 1.6, resulting in 776.9 kg of MMH and 1243.1 kg of NTO. Using the densities of MMH (870.1 kg/m^3) and NTO (1431 kg/m^3), the volumes were calculated for the tanks. These volumes are 0.8929 m^3 for MMH and 0.8687 m^3 for NTO. The exact shapes of the tanks are given in the Structures

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subsystem portion of this report.

The sizing of the first phase of the propulsion system was accomplished once again by using the rocket equation. This time the final mass included the instrument bus, payload, aerobrake shield, and the second phase of the propulsion system. This mass was calculated as approximately 8675 kg. The specific impulse for this phase is 460 sec. According to MMPC and Aerobrake subsystems, the total delta-v for this phase is 4.473 km/s. This gives us a total propellant mass of 14,720 kg. The oxidizer-to-fuel ratio for this combination is 6, resulting in 12,617 kg of liquid oxygen and 2103 kg of liquid hydrogen. This translates into 9.7283 m^3 of liquid oxygen and 26.1336 m^3 of liquid hydrogen. A tankage factor of 0.07 was used to determine the mass of the tanks as well as the engines to be 1036 kg. As one source indicates, "For large chemical propulsion stages...the weight of the engines and control systems can be included in the massless parameter" (Babb 13). As indicated in Fig. 2, this phase requires the use of two pumps. The helium tanks are once again required to prevent vaporization of the liquid fuels.

In summary, the propulsion system is divided into two phases: the first using liquid oxygen and liquid hydrogen, and the second using MMH and NTO (pressure-fed). The choice of the first phase was based on the high specific impulse for the large delta-v, as well as reliability. The choice of MMH/NTO for the second phase was based on the fact that both are storable in space for extended periods of time. The tankage of the first phase is jettisoned after Mars capture.

Power Subsystem

The Power Subsystem of Project ACRONYM is required to provide an uninterrupted supply of power to the spacecraft. Due to the rather low power requirements of the spacecraft, the choice of a power source was between RTG's and solar arrays; any other system would weigh too much and would not be feasible. The biggest advantage of RTG's is the constant, steady supply of power that is independant of distance from the sun. However, our mission will never be more than 1.54 AU from the sun, and the distance from the sun, therefore, will not be a major problem. Also, since the political situations in our country give top priority to defense, it is highly unlikely that any RTG's will be available for our mission. Therefore, solar arrays were chosen to supply power to ACRONYM.

During its orbit about Mars, the spacecraft will at times be in Mars' shadow. For this reason, the spacecraft is required to carry rechargeable batteries. The choice of batteries was between Nickel-Hydrogen and Nickel-Cadmium. The biggest advantage of using Ni-H_2 batteries is a longer lifetime at the same depth of discharge (DOD). However, for a mission lasting less than ten years, Ni-Cd batteries are often chosen for their low cost. Although they will operate at a lower DOD, Ni-Cd batteries will provide sufficiently for our mission.

According to one source, "Bus voltages of 100 to 120 V are being designated for some spacecraft now, and will probably be quite common in the late 1980's and beyond" (Agrawal 367). Since the choosing of bus voltages is often based on a desire to use off-the-shelf hardware, a bus voltage of 110 V was chosen. Due to the high bus voltage, a regulated bus is needed. This is especially true when the spacecraft comes out of an eclipse; since the cells are cold at this point, the bus voltage is expected to jump extremely high (perhaps as high as 200 V). In order to eliminate single point failures, a dual bus was chosen. Figs. 3 and 4 give visual descriptions of the power system for Project ACRONYM.

In order to size the solar arrays and batteries, power requirements were obtained from each subsystem. The requirements for PPS are listed along with requirements from the other subsystems in Table 1. The solar arrays were sized for a total power requirement of 600 Watts. Power fluctuations are controlled by the regulator shown in Fig. 3. The total power required was obtained by using the following equation:

$$P_T = \frac{P_L t_o}{(\eta_o \eta_r \eta_t)(\eta_a \eta_b \eta_c) t_s} \times \frac{(\eta_a \eta_b \eta_c) t_s + t_d}{t_s + J t_d}$$

where P_L = power to load = 600 Watts

t_o = time of orbit = 24.623 hours

t_d = time in dark = 1.2879 hours

t_s = time in sun = 23.3351 hours

J = power in dark/power in sun = 1

η_t = solar array transmission efficiency = 0.98

η_c = charger efficiency = 0.95

η_b = battery efficiency = 0.80

η_d = battery-to-regulator efficiency = 0.97

η_r = regulator efficiency = 0.91

η_o = distribution efficiency = 0.98

This gives us a total solar power requirement of 740 Watts. Assuming a 30% array degradation over the life of the spacecraft, a beginning-of-life power requirement of 1054 Watts is obtained. The area of the solar array is calculated by using the following equation:

$$P_{BOL} = S \times C_r \times e \times A \times \left[1 - \alpha(T-25)\right]$$

where S at Mars = 575 W/m^2

C_r = solar cell packing factor = 0.88

e = solar cell efficiency = 0.12

α = efficiency drop = 0.005

T = maximum operating temperature = 50 °C

This yields an area of 19.84 m^2. We size the solar arrays at 20 m^2, and, at an areal density of 2.40 kg/m^2, we obtain a mass of 48 kg. The solar panels are designed to be retracted during the aerobraking sequence and during some orbit maneuvers. Details of this feature may be found in the Structures subsystem portion of this report.

In order to size the batteries, a depth of discharge of 40% was chosen. This translates into a maximum of approximately 5000 discharge/recharge cycles for Ni-Cd batteries, or more than ten years of geosynchronous orbit. Given that a Ni-Cd cell has an energy of 30 WH/cell, we can calculate the required stored energy by:

$$\text{Stored Energy} = \frac{P_t t_d}{DOD}$$

where P = 740 Watts. This results in a stored energy of 2383 WH. Now, the number of cells required can be calculated using:

$$\text{No. Cells} = \frac{\text{Stored Energy}}{\text{WH/cell}}$$

This yields 80 cells. The battery capacity can now be calculated by using:

$$\text{Battery Capacity} = \frac{P_t t_d}{DOD \cdot V}$$

which gives C = 21.66 hours. From Marshall Space Flight Center data, we know that Ni-Cd batteries have an energy density of approximately 30 WH/kg. From this, the mass of the batteries can be calculated as 79.4 kg. However, to avoid single point failure, Project ACRONYM uses completely redundant batteries. Thus, it carries 160 cells and 158.8 kg of batteries.

In summary, the power system uses solar arrays and Ni-Cd batteries to supply power to the spacecraft. No single point failures are possible with the configuration designed for this system. I feel this system will be able to provide power for much longer than the four years of orbiting required.

Problem Areas

Although a thorough design study was done, some problems remain unsolved. One of these problems is heating of liquid hydrogen and liquid oxygen during the trip to Mars. Another problem is the repeated deployment and retraction of the solar panels.

Although the liquid fuel tanks are protected by the shadow of the aerobrake shield during the trip, it is difficult to say what the temperature will be in the shade. If the temperature exceeds the boiling point of hydrogen, the system will fail. One possible solution may be to protect the tanks with insulation. However, even the best insulation is not perfect. Perhaps, a refrigeration system is possible.

Aerobraking is a relatively new idea, and new problems will arise from it. One such problem is the repeated retraction and deployment of the solar panels. To date, all solar panel deployment schemes have been designed to deploy the panels once. I found no record of any spacecraft that was required to retract its solar panels so many times. I know of no way to predict how aerobraking will affect the solar arrays other than actual testing.

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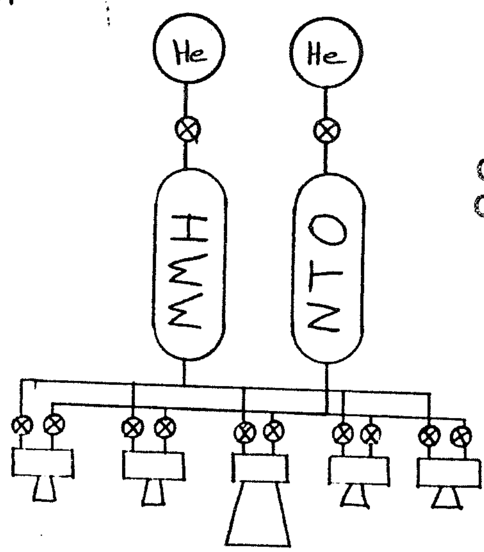

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Fig 1: Schematic diagram of second phase of propulsion system.

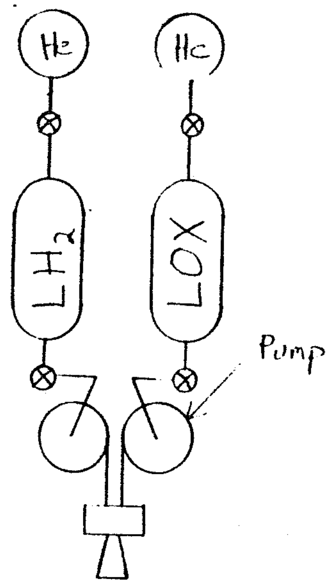

Fig 2: Schematic diagram of first phase of propulsion system.

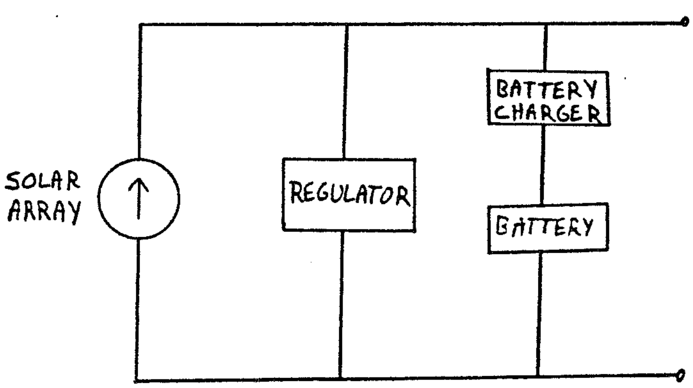

Fig 3: Power system circuit diagram.

Table 1: Power Requirements

Command and Communication Control	214 Watts
Science Instrumentation	100 Watts
Attitude and Articulation Control	209 Watts
PPS--4 valve actuators	40 Watts
solar panel actuators	20 Watts
	583 Watts

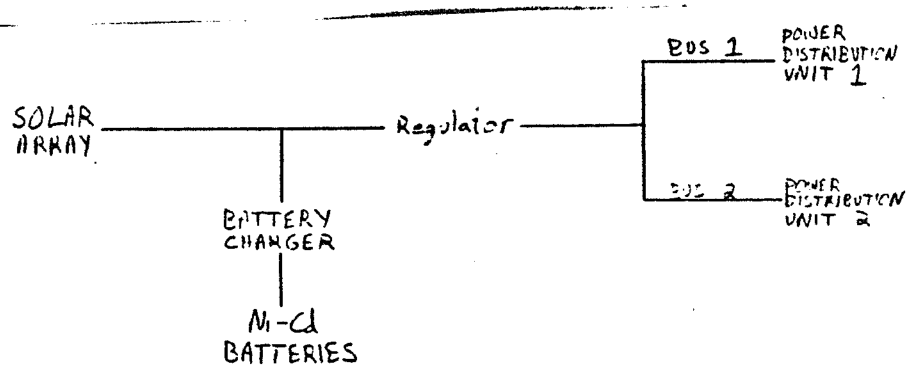

Fig 4: Flow of power in the power system.

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Attitude and Articulation Control Subsystem (AACS)

Project ACRONYM

by Russel S. Wenzel

Group #5

Requirements

During the lifetime of ACRONYM, the Attitude and Articulation Control Subsystem (AACS) must be able to:

- Send telemetry to, and accept commands from the Command, Control, and Communications (C3) subsystem.
- Request power from the Power and Propulsion Subsystem (PPS).
- Control outputs that affect the spacecraft attitude, alter the scan platform or antennae pointing positions and actuate any valve or thruster necessary for attitude adjustment.
- Receive and monitor the sensor inputs from the attitude reference system components such as gyros, accelerometers, star trackers, star scanners, or sun sensors.
- Receive and monitor the various actuator position encodings.
- Receive and monitor the various valve actuation signals from the fuel and oxidizer lines, and the thrusters themselves.

Mission Phase Breakdown

From the perspective of the AACS, the entire ACRONYM contains three separate craft: the EMTV (Earth-Mars Transport Vehicle), the satellite or instrument bus and the PL (Payload Vehicle). Hence three different attitude adjustment scenarios are presented.

Control System Selection

EMTV

It was decided early on that the EMTV would be three axis stabilized. Table 1 shows some essential merits of why this was so. It should be pointed out that to offset the continual one-sided solar heating problem that will occur during the transfer flight, a slow specified roll rate was considered as a possibility. Due to the added complexities, however, and the existence of active and simple passive cooling techniques, the idea was soon discarded.

Because of the EMTV's large size, it became apparent that the introduction of a reaction wheel system would be a considerable venture, even though it might save fuel. Therefore, an all thruster system utilizing a combination of 38 variously sized thrusters (Ref Table 6, 9), with full maneuvering redundancy was designed. The expected pointing accuracy, by extrapolating on the abilities of current technologies, is 2 mrad, which easily exceeds the communications requirements of 8.108 mrad, due to its dispersion angle (Ref Table 10).

The full EMTV AACS fuel budget, including pre- and post- aerobrake turn manuevers, is given in Table 4. The total known consumption is 49.362 kg. In-transit manuevers were estimated by the "Lembeck Approximation": Enough fuel to turn the SC around once a day. Other rates for take-off and aerobraking were purely estimated.

Satellite/Instrument Bus

Since the bus' computer essentially commands the whole mission, and the EMTV is three axis controlled, the satellite, logically, should be also. Design considerations verified that assumption. Due to its integration within the EMTV, the only feasible means of solar array deployment was by the addition of panels. This was a logical choice as only half the craft would be sunlit anyway. So the panels became inherited in the satellites design, and immediately spin and dual-spin stabilization techniques were discarded.

To save thruster fuel, and to look into the possibility of more accurate and stable pointing, various other methods of three axis

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stabilization schemes were also explored. Table 2 shows the various design trade-offs that were considered. The OPTRANSPAC placement on the sides of the satellite, for redundancy reasons, dictated that the face of the satellite remain as normal to Mars as possible. The requirement of orbit transferability and maneuverability also dictated transfer simplicity. The results were that a 3 axis-momentum biased system was chosen. However, this requires some ground sensing, and thus we will assume that the Mars base is capable of detecting and controlling pitch/roll of the satellite.

A major stability problem would have occurred in any of the previous 3-axis stabilization methods, had it not been for the addition of an IPPACS-like (Integrated Platform Pointing and Control System) stability platform utilizing the technologies of the MMII mission. The science subsystem required a strict 7.5 microradian/sec jitter rate maximum. The bus alone would not be able to meet this requirement, but this new platform offers that capability. By incorporating highly accurate movement sensing on the platform itself, and mechanically offsetting the potential problem by microactuating the platform, jitter rates less than 1E-7 radians may be achieved. In essence then, the platform becomes decoupled from the movements of the bus in 2 DOF.

The bus system itself is rated to have a mere 2 mrad orientation inaccuracies. When earth and mars are diametric with respect to the sun, the focal point of the communications laser with earth would be offset some 117.4 earth radii! Fortunately, the laser that OPTRANSPAC utilizes has a 8.108 mrad dispersion angle: twice the amount necessary to offset this inherent problem.

A propellent budget for the Satellite is given in Table 5, with reference to Figure 1. A couple of analytic results become obvious during the transfer flight development. First, by doing a Hohmann orbit transfer first, and then an equatorial orbit transfer, a delta v savings of about 0.85 km/s is made in comparison to doing the procedure in reverse. Secondly, by requiring the face of the satellite to be normal to the surface of Mars, the satellites arrays will be offset some 25deg10min (the inclination of the Martian equator to its orbit) from the direct rays of the sun. This factor is taken into account when array sizing is analyzed (See PPS).

Finally, it will be assumed that the reaction wheel inside the satellite can be completely desaturated and shut off. This assumption is necessary to make orbit transfers utilizing spin ups along the principle moment of inertia easier.

Payload Vehicle (PV)

The Articulation control for the PV is substantially cruder than that of the EMTV or bus. After separating with the instrument bus in lower orbit, the PV is still dependent upon the Satellite's computer for AACS instructions. A low-gain antenna communications link is provided for continuous contact all the way to landing. The larger 20lbf engines kick in, producing a minimum delta v of 37.6 m/s to put the PV into an elliptical orbit with a perigee just touching the fringe of Mar's thin atmosphere. The primary purpose of the reentry AACS is to make sure the shell points in the direction of travel, so atmospheric friction won't fry up the Mars aircraft cargo. This will be accomplished by implementing a crude guidance system somewhat similar to the V2 rocket technology, i.e. a gyro senses a pitch, yaw or roll, and actuates thrusters located upon the PV. Communications with the satellite computer will then be required for final entry actuation. Thus, until the go ahead is given, PV will have to maintain stability and attitude with this system. For details, refer to the Aerobrake section.

System Configuration

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The Layout of the various AACS components for project ACRONYM is given in figure 3. During the transfer flight from Earth to Mars, prior to the aerobraking sequence, all the sensor and control feeds will be sent, in their analog form, directly to the computer in the bus section. Even after the tank section is dropped, the PV is still dependent on the bus computer for thruster acquisition and charge detonation signals. Only when the PV has gone well into entry sequencing, is the PV's primitive articulation control system on its own. Thus, all actuation signals from the Tank or PV sections must be sent via umbilical cords or low - gain com links.

During the various phases of the ACRONYM's mission, many preprogrammed modes will have to be implemented in the AACS portion of the computer. The particular mission modes are given in Table 7. Each control mode really equates to a basic constituent of the total AACS system, and is broken down into the smaller control modes shown in table 8.

Every particular AACS component chosen for the ACRONYM mission is shown in Table 9. The sun sensors were chosen and placed to provide a 360 degree field of view without array hindrance and capable of narrowing down placement to accuracy good enough for gyro updating. Due to their proven reliability, and simple design, no backups have been utilized.

Furthermore, due to the lack of a rotating platform during the transfer mission, an improvised solution was developed to allow star sensor placement. By altering the structure of the OPTRANSPAC platform, a star tracker will be placed next to and simultaneously be pointed with the laser itself. This system provides equivalent pointing capabilities of a scan platform, and yet induces no additional clutter to the bus surface. The star tracker itself will be assumed to have the capabilities of the planetary ASTROS trackers detailed in Table 2, and thus help give the entire mission the expected pointing accuracy given in table 3.

To sense pitch, roll and yaw rates, the ACRONYM will utilize the latest in gyroscopic technologies. The FORS (Fiber Optic Rotation Sensor) should be well proven by 1998 on the Mariner Mark II missions, and offer a considerable improvement in resolution, drift rates, power consumption and mass. It offers full redundancy in 3 -axis, and has a phenomenal meantime between failures of 10 years- easily meeting both our lifetime requirements and stability sensing requirements for the communications system.

The thruster system, for providing corrective torques, is specified by type in Table 9. During the transfer, up to the point of fuel tank separation, the PV and tank thrusters provide full redundancy for all corrective maneuvers. Table 6 shows all the thruster variations for pitch, roll and yaw adjustment.

Thruster placement was dictated mostly by the Solar arrays. In order to prevent damage to the panels, all satellite and PV thruster quads were offset 45 degrees. When this occurred, however, we found 4 of the satellite's thrusters blasting square away on the OPTRANSPAC platform. Fortunately, they were determined to be redundant, and thus struck from the final design. The other four counterparts, also were found to cause problems of plumage on the array. It was then determined to lengthen the array stem a minimum of 62 cm, and then reshape the stem to a V-Shape, to totally throw out any thrust reversing effects.

Sizing was done by attempting to equalize the moment contributions by each thruster about the center of mass for a pitch axis rotation like the 180 degree turnabouts for aerobraking. All the PV's thrusters were initially assumed to be the NASA 5lbf standard, and the other thrusters were sized from it. Given the results from INERT for trajectory and aerobraking modes, the corresponding Sizes were then found. The exceptions are PV R&L 5and6, as well as SAT R&L 3,4,7,8. (Ref Table

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6). These PV' were deemed necessarily larger for orbit delta v adjustment and insuring stability during reentry. The SAT exceptions were sized for separation quickness.

As noted on Table 9, the OPTRANSPAC contains its own platform pointing mechanism, similar to an observatory's 2 D.O.F. characteristics. It too, will contain a momentum compensation device, so no appreciable torque should be felt by the EMTV or the bus. Actuation is done digitally straight from the computer.

The solar array actuators were sized and chosen to grant both controlled and continuous rotation characteristics for three mission operating environments: transfer, science orbit, and stationary orbit. The rotation rates required were .052 and .004 deg/sec for the science and stationary orbits, respectively. Design of such an actuator is possible, especially in the advent of a high precision stepper driver as the previously mentioned microstepper motor for the platform. It can be easily procured before 1998, and should consume less that 5W and be about 5-10 kg, in accordance with todays standards.

The last component selected was the reaction wheel for the satellite. Its main purpose was viewed to insure the satellite face would be normal to mars, so both OPTRANSPAC lasers could be facing either to earth or mars to insure full redundancy purposes.

The biggest major inhibitor of maintaining continuous alignment, since both OPTRANSPAC platforms, and the stability platform contain their own momentum compensation devices, was the constant torque caused by the solar array actuators. As they spin around at a maximum rate of 906 microradians/sec in low orbit, they impart a frictional torque to the Satellite, which is trying to spin at exactly the opposite spin rate. Sizing was then done assuming a 0.1 sec momentum drain time exists on the wheel, while a constant torque is imparted. The angular impulse had to equal twice that of the angular momentum offset of the array actuators. The average torque was determined to be 0.145 Nm, and off the shelf hardware was searched for to meet this demand. It was found that the Sperry reaction wheel met this description rather well, and it was then introduced to the final design.

Other Considerations

Other possible problems confronted by the AACS was how to separate the PV-Bus combination so remote sensing could begin. In the case of tank separation of the EMTV, it was decided to separate by explosive charges at the joint, and letting the main satellite motor blow it away during the aerobrake process. For the Bus/PV separation, it is planned to rotate the ship 180 degrees in opposition to the its motion, fire the pyrotechnic charges, and then back off the bus 15 meters by using four of its 5lbf articulation thrusters. The acceleration profile, along with fuel sizing configurations is shown in figure 4. The process is slow, but relatively effective, as very little orbit perturbation is anticipated. The PV will then have to wait close-by, maintain its attitude, and wait for the call from the Satellite computer for the go ahead to kick in its engines.

Problem areas

As the final design numbers slowly came in, it became apparent that not everything is going to work as nicely as planned.

First, the center of mass of the bus was not located at the 1.0 meter mark through its width. In fact, it was offset 25 cm forward. This would appear to be due to the addition of the stability platform. Secondly, cross products of inertia exist throughout the mission lifetime. This does not make fuel consumption very low, nor thruster placement always ideal. Finally, the placement of the OPTRANSPAC (and hence the star trackers also) behind the shield inhibits direct identification of mars until a 180 degree turnaround is established, as well as narrowing the FOV of the sun sensors. This

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placement problem could quite definitely present a problem for overall tracking ability on the way to Mars.

However, the technology is there, and the overall ability of the AACS system to handle the mission is quite good.

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Table 1 EMTV Control System Trades

Scale 3:Positive 2: Neutral 1:Negative

Criteria	3 Axis Active	Spin Stabilized
Simplicity	Relatively (3)	Complex (1)
Pointing accuracy	Good (3)	Good (3) Assuming Dual Spun
Proven Interplanetary Technology	Yes (3)	No (1)
Procurement Costs	Fair (2)	High (1)
Totals	11	6

Table 2 Satellite 3-Axis Control System Trade-offs

Weight system: 5 = positive, 3 = neutral, 1 = negative

	All Thruster	Momentum Biased	CMG system
Orbit Transfer & Shut-down Simplicity	4	3	2
Planet Facing	3	4	3
Fuel Savings	1	4	5
Stability	3	4	5
Dynamics Simplicity	5	4	3
Long Term Reliability	3	5	5
Totals	19	24	23

Table 3 Orbit Data

Kepler's 3rd Law: $T = 2\pi\sqrt{\frac{r^3}{\mu_{♂}}}$
$\mu_{♂} = 4.305E13\ m^3/s^2$

Low Orbit (Science Monitor Orbit)
height = 350 km : r = 350km + 3393km = 3743 E3 meters
T = 6,934.63 sec = 115.577 min
w = 906.05923E-6 rad/s = 51.913 E-3 deg/sec
v = rw = 3.39138 km/s

Aerostationary Orbit
height = 17,069.83 km : r = 20462.83 E3 meters
T = T(Mars) = 24hr37'22.668" = 88,642.2 sec
w = 70.882 E-6 rad/sec = 4.016 E-3 deg/sec
v = rw = 1.45045 km /s

Table 4 EMTV AACS Fuel Budget
Earth Escape to post-Aerobrake Phases

Manuever	Max Magnitude	Est. Fuel Mass (kg)
Launch Attitude Maintenance	90 secs at 3.14 rad/15s *	17.17
Journey Attitude Maintenance	200.8 days at 360 deg/day	0.673
Pre-deltaV 180 Turnaround @ Mars	180 deg/60s *	2.112
Pre-Aerobrake 180 Turnaround	180 deg/60s *	1.422
Aerobrake Attitude Maintenance	7 times at 180 deg/60s *	25.0
Post-Aerobrake, pre-separation turnaround	180 deg/60s *	2.985
Totals		49.362 kg Vol = 0.0567 m^3

* Estimated turning rates

Fuel Mass estimations were calculated by the following derivation:

$$M = \dot{H} = I\dot{\omega} = Fl$$

$$l\int_0^{t_{FIRE}} F\,dt = I\int_{\omega_i}^{\omega_f} d\omega = I(\omega_f - \cancel{\omega_i}^{0})$$

ASSUME FOR THE THRUSTER TIME OF FIRE, t_{FIRE}, $\int_0^{t_{FIRE}} F\,dt \approx F t_{FIRE}$
BUT, FOR A THRUSTER: $F = \dot{m} g_0 I_{sp}$. HENCE, $\int_0^{t_{FIRE}} F\,dt = \dot{m} g_0 I_{sp} t_{FIRE}$. NOW,
ASSUME $\dot{m}$ FOR FUEL CONSTANT DURING DISCHARGE, THUS $\dot{m} t_{FIRE} \approx \Delta m$.
SUBSTITUTING,

$$l\left[\Delta m\, g_0 I_{sp}\right] = I\omega_f$$

$$\Delta m = I\omega_f / g_0 I_{sp} l$$

Hence, the maximum amount is calculated by turning along all three axis:

$$\Delta m = \left(\frac{I_{xx}}{l_x} + \frac{I_{yy}}{l_y} + \frac{I_{zz}}{l_z}\right)\frac{\omega_f}{g_0 I_{sp}}$$

Moments of Inertia, and lever arms, , are given by INERT data for various mission phases (see Structures). g_0 = 9.8 m/s², I_{sp} = 340 secs.

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Table 5 Satellite Fuel Budget

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Aerobraking complete, 350 km orbit height established

Manuever	Maximum Magnitude	Fuel Mass (kg)
Separation	0.673 m/s @ 1min	.4118
N-S Station Keeping	11.8 m/s [a]	<< 1
E-W Station Keeping	Unknown [b]	<< 1
Attitude Maintenance	19 Days	< 1
Wheel Desaturation	4.15 kgm^2/s [c]	0.00083
Spin Up	30 rev/min [d]	1.76
Hohmann Transfer 3,743km->20462.8km	1.663 km/s [e]	705.0
Spin Down	30 rev/min	1.65
Reorientate	192deg35min @ 60 secs	0.035
Spin Up	30 rev/min	1.65
Equatorial Orbit Transfer	0.632 km/s	188.0
Spin Down	30 rev/min	1.62
Reorientate	90 deg @ 15s	0.032
N-S Station Keeping	4.9 m/s [a]	<< 1
E-W Station Keeping	Unknown [b]	<< 1
Attitude Maintenance	4 Years +	8.0
Residual		10.0
Total		918.159+ kg
		Vol = 1.05 m^3

Footnotes for Table 5:

a) Reference Agrawal delta V = 2 V sin(Theta/2)
= 2 V Pi/180 x i deg
where i is the maximum allowable orbit inclination. Letting i = .1 deg, and substituting the values of V for the two orbit heights, one obtains the answers given above.

b) Reference Agrawal

Since the Magnetic Fields of Mars are not known to any sufficient degree of accuracy, J_2 -- a vital parameter for the station keeping equations, is not either, and hence only an estimate may be applied.

c) Wheel Desaturation

Knowing the Angular Momentum of the wheel, and utilizing M=dH/dt=Fl, Impulse = m g_o I_{sp},

$$m = H / (g_o I_{sp} l)$$

In this instance, H = 4.14765 kg m^2/sec, g_o = 9.8 m/s^2, I_{sp} = 340 s and l= 1.5 m gives our result of .0083 kg

d) Satellite Spin Up

Derivation Similar as c), except H = I x angular rate. The value of I is found using INERT (see Structures)

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e) Hohmann Transfer

$$\Delta V = \sqrt{\mu_r 2\left(\frac{1}{r_i} - \frac{1}{r_i + r_f}\right)} - \sqrt{\mu_r / r_i} + \sqrt{\frac{\mu_r}{r_f}} - \sqrt{\mu_r 2\left(\frac{1}{r_f} - \frac{1}{r_i + r_f}\right)}$$

Letting initial radius = 350km high = 3.743 E 6 m
final (areostationary orbit) = 20.46283 E6 m
and Mu = 4.305E13 m^3/s^2
one obtains
delta V = 1.663 km/s

The transfer time is given by $T = \pi \sqrt{\frac{(r_i + r_f)^3}{8\mu_r}}$

and in this case,

T = 20,156.65 s or 5.599 hrs

Table 6: Thruster Operation Modes

EMTV

Function	Combination
Roll	
+	PV(R3&L8);PV(R4&L7)
-	PV(R7&L4);PV(R8&L3)
Pitch	
+	TANK(3&4);PV(L1,7&L2,4)
-	TANK(5&6);PV(R1,7&R2,4)
Yaw	
+	TANK(2);PV(R1,3&L1,3)
-	TANK(1);PV(R2,8&L2,8)

PV & Bus Combined

Function	Combination
Roll	
+	PV(R3&L8);PV(R4&L7)
-	PV(R7&L4);PV(R8&L3)
Pitch	
+	SAT(R1&R2);PV(L1,7&L2,4)
-	SAT(L1&L2);PV(R1,7&R2,4)
Yaw	
+	SAT(L2,5&R2,6);PV(L1,3&R1,3)
-	SAT(L1,5&R1,5);PV(L2,8&R2,8)

Instrument Bus

Function	Combination
Roll	
+	SAT(R3&L3);SAT(R8&L8)
-	SAT(R7&L7);SAT(R4&L4)
Pitch	
+	SAT(R3&R4);SAT(L7&L8)
-	SAT(R7&R8);SAT(L3&L4)
Yaw	
+	SAT(R5);SAT(L2)
-	SAT(R2);SAT(L5)

PV Alone

Function	Combination	Function	Combination	Function	Combination
Roll		Pitch		Yaw	
+	PV(R3&L8);PV(R4&L7)	+	PV(L5&L6)	+	PV(R5&L5)
-	PV(L3&R8);PV(L4&R7)	-	PV(R5&R6)Yaw+PV(R5&L5)	-	PV(R6&L6)

Where ; is an either/or conjunction and the , is a conjunction

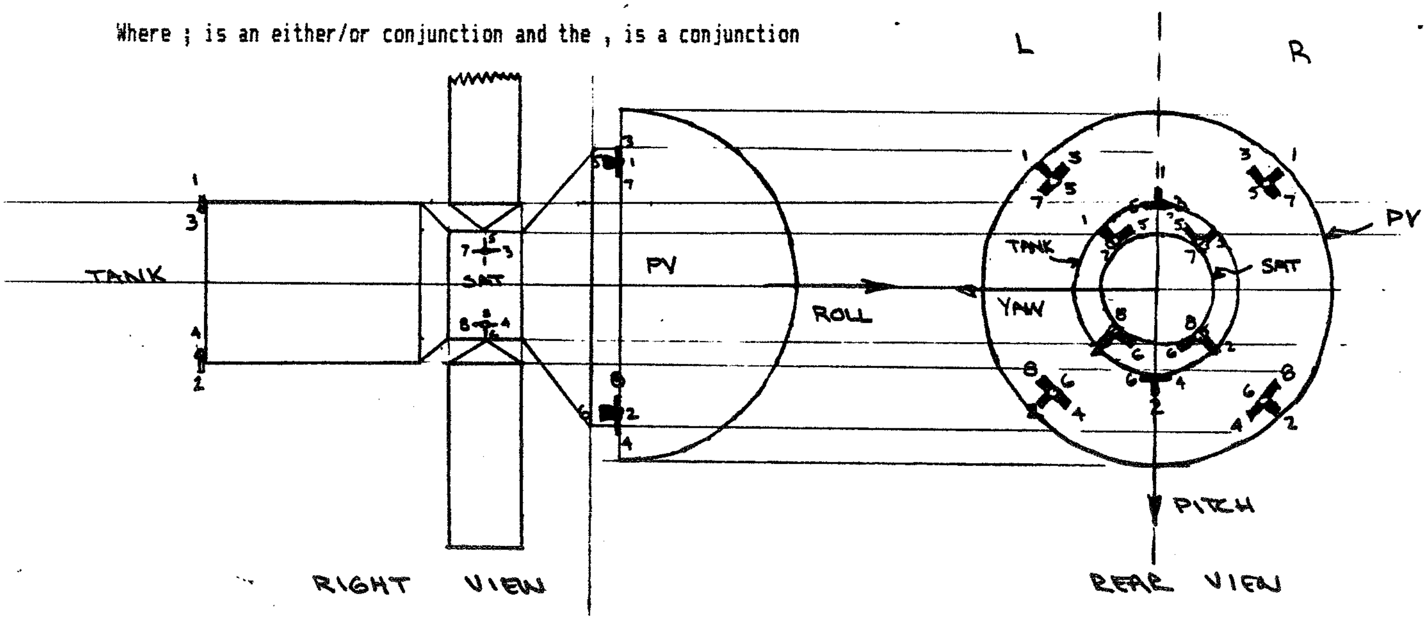

Table 3 AACS Mission Modes

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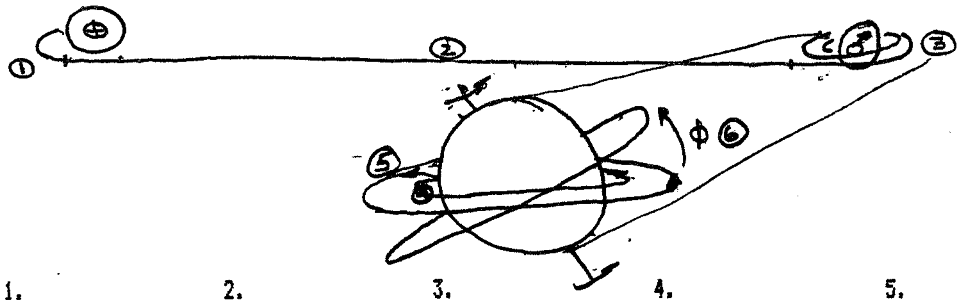
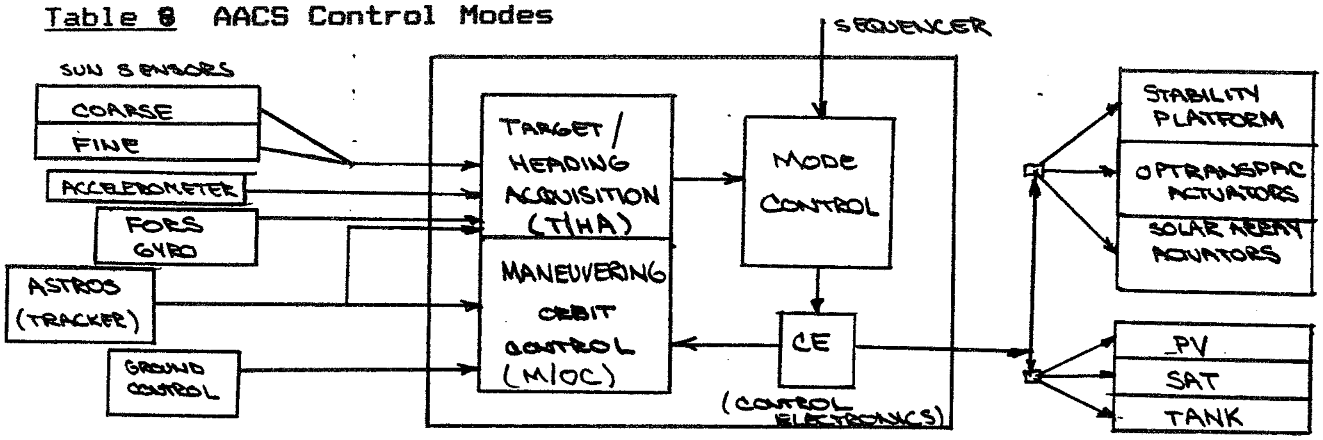


1. Earth Escape Modes	2. E-M Transfer Modes	3. Aerobrake Modes	4. Science Scan Modes	5. Areo-Assault Modes	6. Equatorial Modes
Position Update	Position Update	Position Update	Position Update	Array Retraction	Spin Up
Stabilize	Health Update	Reorientate	Reaction Wheel+	Desaturation	Timing
Health Update	Gyro Update	Stabilize	Platform Activate	Platform Shunt	Despin
	Array Pointing	Gyro Update	Array Pointing	Spin Up	Position Update
	Comm Pointing	Health Update	Comm Pointing	Timing	Reorientate
		Array Retraction	Orientate	Despin	Reaction Wheel+
		Stabilize		Position Update	Array pointing
		Array Pointing		Comm Pointing	Comm pointing
		Comm Pointing		Reorientate	Orientate
		Separation			Platform Activate

Type	Frequency	Purpose	Control Modes Called
Position Update	Continuously	To Determine the current position of the S/C and compare it to the preprogrammed data.	Sun Acquisition, Mars Acquisition, Star Acquisition, Earth Acquisition
Stabilize	As Needed	To insure that the actual path coincides within the ideal trajectory	Fine Pointing, Maneuvering, Delta V trim
Health Update	Continuously	To relay health and trajectory telemetry to earth or mars base	Fine Pointing
Gyro Update	Every 26 hours	To recalibrate the FORS gyro system after natural drift rates overcome the pointing error.	Sun Acquisition, Mars Acquisition, Star Acquisition, Earth Acquisition Fine Pointing
Array Pointing	EMTV: 1 hr Bus: continually	To keep the arrays pointed at the sun	Fine Pointing, Actuation
Comm Pointing	Continuously	To keep at least one OPTRANSPAC communications laser pointed to Earth or Mars stations	Fine Pointing, Actuation
Reorientate	As called for	To turn the EMTV or the Bus through a a specified angle in space.	Manuever, Fine Pointing
Array Retraction	As called for	To retract arrays for next manuever	Actuation
Separation	As called for	To separate the vehicle into parts	Actuation
Reaction Wheel+	As called for	To initiate reaction wheel	Actuation
Platform Activate	As called for	To induce platform self-control	Actuation
Orientate	Continuously	To maintain satellite normal to Mars	Fine Pointing, Manuever
Desaturation	As called for	To slow down a rapidly spinning wheel	Actuation, Fine Pointing
Platform Shunt	Once	To shut off stability platform during transfer	Actuation
Spinup	As called for	To spin up principal axis for transfer	Actuation
Timing	Once/orbit trans	To wait for the duration of transfer	none
Despin	As called for	To despin principal axis after transfer	Actuation

Table 8 AACS Control Modes



Control Modes (All mode controls originate in Mode Control box)

Type	Process
Sun Acquisition	Input received from TH/A from digitally converted coarse sun sensor; one is shunted; Call made to CE for maneuvering till null position achieved; process repeats for fine sensor; Attitude is now reset in THA memory
Earth Acquisition	Input received from TH/A last call; New position computed by updating all previous gyro and accelerometer calls; OPTRANSPAC call to station verifies, or controlled search initiated by CE and OPTRANSPAC actuator pointing.
Mars Acquisition	Input received from TH/A last call; New position computed by updating all previous gyro and accelerometer calls; OPTRANSPAC call to base verifies, or controlled search initiated by CE and OPTRANSPAC actuator pointing.
Star Acquisition	Input received from TH/A last call; CE calls OPTRANSPAC actuators to point trackers to last known position; Position is verified unless no signal to TH/A is returned; MC then activates CE to fire a controlled loop scan search of nearby area until positive signal returned; Position reset in Memory.
Maneuvering	Input received from either TH/A receiving unacceptable slew rates from gyros, or from pre-timed manuever; Call made to CE for correct thruster actuation; position updated on M/OC and actuation signals sent to correct thrusters.
Fine Pointing	Input received from TH/A last call; Call sent to CE for correct actuator selection; M/OC receives updates from latest tracker call; upon unacceptable variance CE calls actuators or thrusters to either alter current condition or maintain it; process repeats if necessary
Actuation	Input received from pre-timed manuever in program memory; call made to CE for correct actuation type and M/OC monitors feedback signal and updates status.
Delta V Trim	Input received from TH/A monitoring digitally converted gyro and accelerometer rates, or from preprogrammed sequences; call made to CE which sends M/OC the requested actuation; upon update and verification; actuation signal are fedback to the main engines.

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Table 9 Selected Components and Placement

Component	Number	Placement	Performance	Mass	Power	Dimensions
FORS	2	Center Of Bus Stability Platform	Drift Rate: 2E-4 deg/hr Rate Noise: 1E-5 deg/sec Ang. Resol: 5E-3 arc-sec BTBF : 10 years	10 kg	<10 W	1000 cub. in.
Adcole OAO-B ATS Coarse Sun Sensor	2	Perimeter of bus 90 deg offset of solar arrays both sides	FOV : 180 deg solid angle Accuracy: 2 deg at null Two axis Analog System	0.18 lb	None	1.9x1.9x1.3 in
Adcole SFPS (Standard Fine Pointing Sensor)	2	Perimeter of bus 90 deg offset of solar arrays next to OAO-B's	FOV : +/- 2 deg/ axis Accuracy: 5 arc sec Generates sun position in two orthogonal axis Analog System	Head: .88 Elec: 1.76 Head Mtg: 3.09 lbs	2.2W	Head: 22 cub in Elec: 91.5 in^3
Planetary ASTROS Star Sensor	3	One on stability platform; One on either OPTRANSPAC module	Sensitivity (Mag): -2 - 6 FOV: 11 deg x 11 deg Drift Rate: <.5 deg/S Three Star simultaneous meas. with internal redundancy	8 kg	11 W	25x16x16 cm
Standard Accelerometer	6	One on each of 3 axis in bus & PV	Fine Sensing Full Redundancy	3 kg	1 W	10x5x10in
Solar Array Actuator	2	Mounted on bus-array connection parallel to pitch	Controlled and continuous rotation rates, 15 Nm Torque, 0.001 deg/step, 0.004 deg/sec- .052 deg/sec continuous rotation	7 kg	5 W	20x20x40 cm
Thrusters	38					
5 lbf	20	Payload Vehicle PV R&L 3-8 SAT L&R 3,4,7,8	Proven reliability	10.0 lbs	10 W	Various
1.7 lbf	6	Tank Clusters Tank R&L 1-6	To be tested			
1.3 lbf	8	Satellite Sat R&L 1,2,5,6	To be tested			
20 lbf	4	Payload Vehicle PV R&L 5,6	To be tested			
Sperry 15 (HEAO) Reaction Wheel	1	Ctr of Satellite Spin axis along array axis	30 ft-lb-sec @ 2000 rpm Torque: 25 in oz (.175Nm)	29.5 lbs	115 W	14x14x7.8 in

Table 9 (cont) Selected Components and Placement

Component	Number	Placement	Performance	Mass	Power	Dimensions
Stability platform:	1	Satellite face	Rigid Graphite Epoxy Jitter Rate: < 1 E-7 rad/sec	50 kg	None	2x2x0.015m
FORS	1	Platform	See above	10 kg	<10 W	1000 cub in
micro stepper	1	behind Platform	6400:1 gear ratio 1 arcsec/step 6 deg/sec slew rate	20 kg	10 W	.2x.1x.1 m
Planetary ASTROS star tracker	1	Platform	See above	8 kg	11 W	25x16x16 cm
Momentum compensator	1	Platform/micro-stepper interface	Able to counter any turning torques of the microstepper or bus jitter, in 2 DOF. With computer.	20 kg	5 W	?
Computer	1	Inside of bus	Memory: enough for science, communications, and AACS needs (about 60k) AACS Input: Analog AACS Output: Digital HALS Implementation	See C3	See C3	See C3
OPTRANSPAC actuators	2	Beneath laser/ tracker dome	Momentum compensated	See C3	See C3	See C3

Table 10 Expected Pointing Accuracy for ACRONYM Mission

Configuration	Accuracy	Notes
EMTV	< 2 mrad with < 20 mrad/sec rates	Based on MMII technology assessment
PV & Bus, Coupled	< 2 mrad	High accuracy still needed.
Bus	< 2 mrad < 1E-7 rad/sec jitter with platform operational	Based on Earth satellite data and MMII estimates

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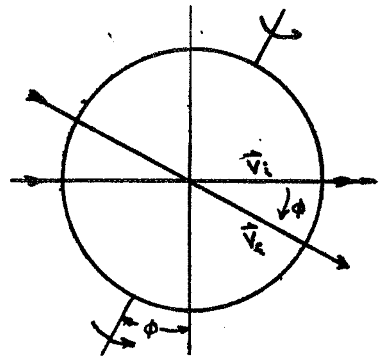
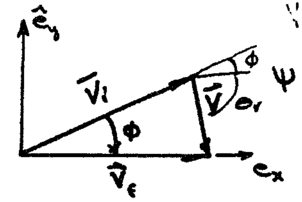
Figure 1 Equatorial Orbit Transfer and Equation Development





$$\frac{V^2}{r} = \frac{\mu_♂}{r^2} \Rightarrow V = \sqrt{\frac{\mu_♂}{r}}$$

$$\vec{V}_i = \sqrt{\frac{\mu_♂}{r}} \langle \cos\phi, \sin\phi \rangle$$

$$\vec{V}_f = \sqrt{\frac{\mu_♂}{r}} \langle 1, 0 \rangle$$

$$\vec{V} = \vec{V}_f - \vec{V}_i = \sqrt{\frac{\mu_♂}{r}} \langle 1-\cos\phi, \sin\phi \rangle$$

$$\Delta V = \|\vec{V}\| = \sqrt{\frac{\mu_♂}{r}\left[(1-\cos\phi)^2 + \sin^2\phi\right]}$$

$$= \sqrt{\frac{2\mu_♂}{r}(1-\cos\phi)}$$

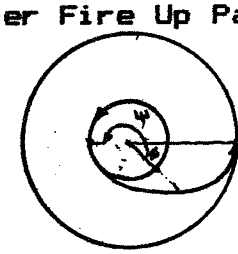
$\phi_♂ = 25°10'$

APPLIED @ $\psi = \phi + \theta_v$

$$= \phi + \tan^{-1}\left[\frac{\sin\phi}{1-\cos\phi}\right]$$

$$= 102°35'$$

Figure 2 Hohmann Transfer Fire Up Parameters



$$\phi = \frac{\text{TRANSFER TIME}}{T_♂} \cdot 360°$$

$$= 80.95°$$

$$\psi = 180° + \phi = 260.95°$$

$$\therefore \psi / \omega_{♂} = 260.95 / 51.91337\text{E-}3°/\text{sec}$$

Transfer must begin 260.95 degrees after last Mars base contact.
This is equivalent to 5026.67 sec = 83 min 46 sec.

Figure 3 AACS Component Placement on ACRONYM (PRE-LAUNCH)

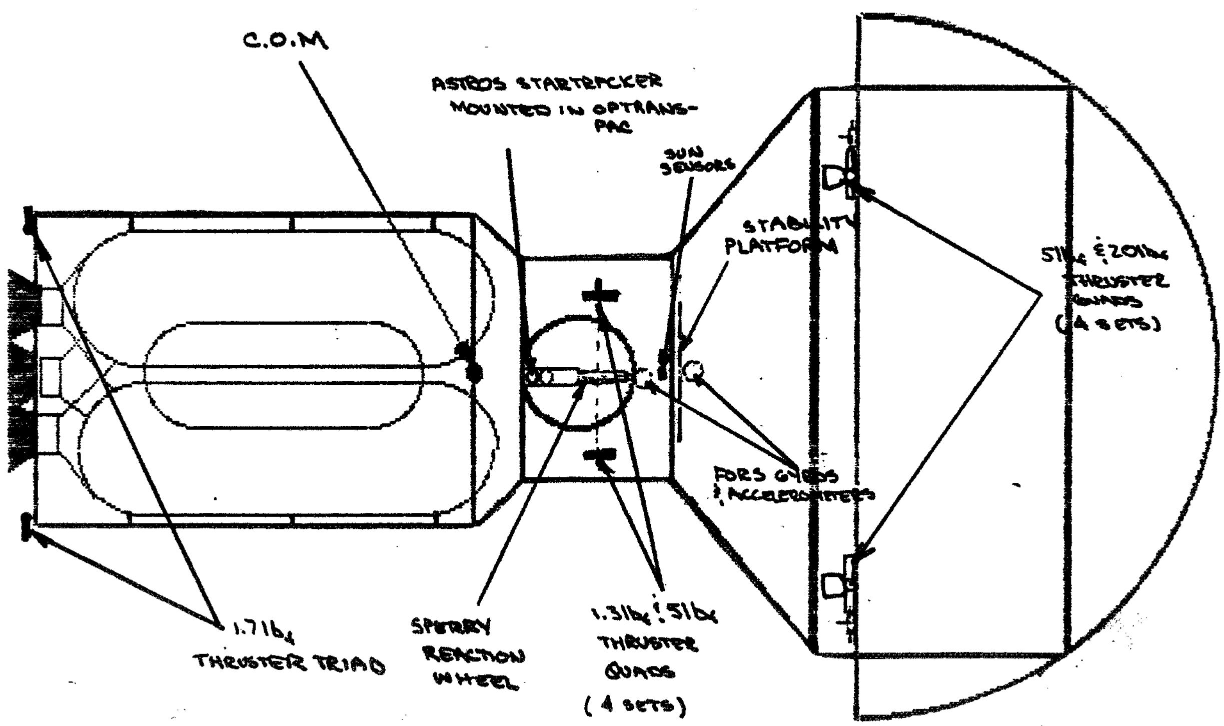
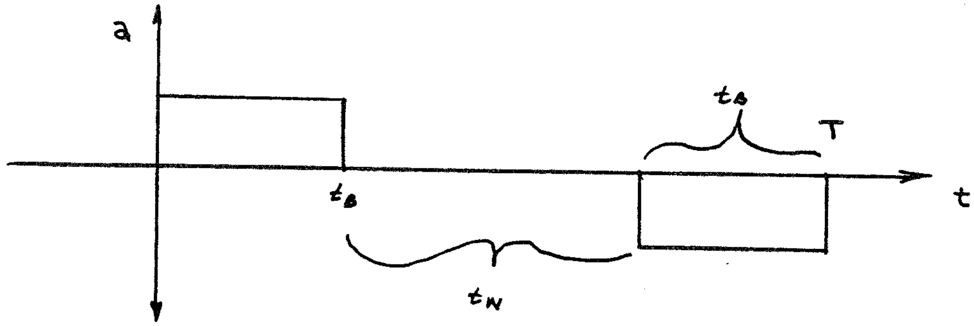


Figure 4 Acceleration Profile and Derivation for Bus Separation



GENERAL PROFILE FOR A CONSTANT THRUST SYSTEM

LET DISTANCE OF SEPERATION BE $d = 15\,m$.

ACCELERATION $\hat{=}$ CONSTANT $= \frac{F}{M} = \frac{4f}{M}$ (4 THRUSTERS FIRE)

$$d = V t_W = (a t_B) t_W$$

$$T = 2t_B + t_W = \text{TOTAL TRANSFER TIME}$$

$$d = a t_B (T - 2t_B) \Rightarrow 2a t_B^2 - aT t_B + d = 0$$

$$\Delta = (aT)^2 - 4 \cdot 2a \cdot d \geq 0 \qquad T \geq \frac{1}{a}\sqrt{8ad} \quad (1)$$

$$t_B = \frac{T}{4} \pm \frac{1}{4a}\sqrt{\Delta}$$

BY REQUIRING $T \leq 60\,s$, (1) REQUIRES $a \geq .03\bar{3}\ \frac{m}{s^2}$

SATELLITE MASS AT THAT POINT IS ~ 2040 kg. HENCE

$$a = \frac{4f}{M} \geq .03\bar{3} \qquad f \geq \frac{.03\bar{3}}{4}(2040) = 17\,N$$

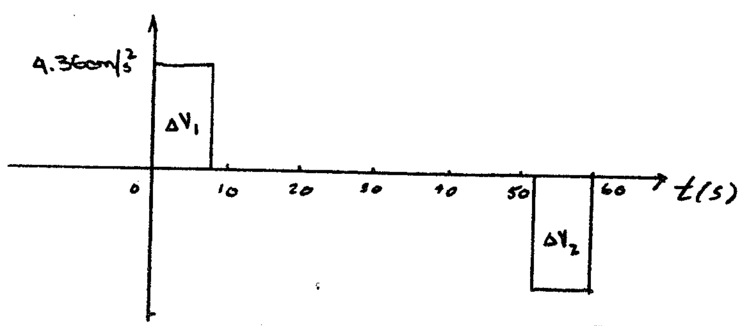
THUS 1.7 lbf OR 1.3 lbf THRUSTERS CANNOT BE USED.

SIZING, THRUSTERS @ 5 lbf (= 22.2 N),

$$a = .0436314\ m/s^2 = 4.36\ cm/s^2$$

$$t_B = 7.7126\ s$$
$$t_W = 44.57\ s$$

HENCE, THE PROFILE IS:



$$\Sigma \Delta V = .673\ m/s$$

$$m_{fuel} = m_{final}\left(e^{\frac{\Delta V}{g_0 I_{sp}}} - 1\right)$$

$$= .4118\ kg$$

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AAE 241 FINAL DESIGN REPORT
STRUCTURAL SUBSYSTEM
GROUP #5
JEFF BRADSHAW

REQUIREMENTS

The structural design for the Earth-Mars Transfer Vehicle (E-MTV) meets the three following requirements:

1. To provide support to components
2. The layout of components meeting specified requirements
3. Thermal control of spacecraft

The components needed to meet the requirements are as follows:

1. Materials
2. Fabrication
3. Thermal control components

Structural Design and Component Layout

This structural layout of the components was primarily affected by the fact that the E-MTV would aerobrake through the atmosphere of Mars. This fact required the structure and the components to be placed safely behind the aerobrake shield. With the primary fuel jettisoned, the placement of the components is shown in figure 2.1. This drawing shows the configuration of the E-MTV during the aerobrake stage. The solar panels are folded in and turned, while the science instruments are safely stored on the plate of the bus. These instruments will be activated immediately after aerobraking when the satellite separates away from the payload bus. This immediate separation was required in order for the science instruments to determine the viability of the predetermined landing site. The intermediate conical shell is positioned between the payload capsule and the relay satellite until their separation. This conical shell was instilled into the design to give the communications system and the solar panels enough distance from the payload capsule

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to operate effectively. The room in the conical shell was used to store the science instruemtns until they were needed and to provide a space for the extra components that were needed on the payload capsule. The E-MTV structural configuration at initial launch is shown in figure 2.2. The components of the mission are listed in detail in table 2.1. The components were placed on the E-MTV according to their requirements. The placements and the masses of the components, along with their body moment of inertias were entered into the inert program to figure the total inertia matrix and center of mass. These inertia tensors are given on the final page of this structural design report.

The payload capsule was designed to utilize the space behind the shield. This was accomplished by storing the aircraft's seven meter airfoil segments with their major lengths transverse to the E-MTV's longitudinal axis. This payload configuration is shown in figure 2.3. The small components of the aircraft are to be stored in two crates. This design gave the E-MTV a center of mass that was just behind the shield after jettisoning the fuel tanks. This fact gives the vehicle inherent stability during aerobraking. The cylindrical capsule is optimal for fitting the payload closest to the shield. The aerobrake shield is supported around the outside of the payload cylinder. The two structures are attached by two rings. One ring is connected on the outside of the payload capsule while the other is implanted in the shield at the proper raduis of 7.6 meters. The two rings separate by pyrotechnic devices, as do the rings joing the other separable structures.

BUS GEOMETRY

The consistent use of the cylindrical shapes of the E-MTV was chosen because of its ease to integrate the different separating structures. The stress analysis is much simpler since the forces of acceleration flow through the outside shells. The analysis of these forces is given in the materials section. The components of the

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mission are mounted on the shells and the plates that enclose the shells. This geometry forms a very simple and practical solution to laying out the components. The layout of the components are shown in figure 2.3 and 2.4. Figure 2.3 shows the different configurations of the components inside and outside of the satellite bus. The components are all specifically required by the other subsystems and can be found listed in table 2.1. The final configuration of the relay satellite is also shown in figure 2.4.

DEPLOYMENT ISSUES

The deployment of the solar panels after the initial velocity change will be completed by actuating an extendable mast which lifts the hinged solar panels away from the bus. These panels are actuated back in near the bus before any velocity changes. They are turned across the main axis of the bus during the aerobrake stage to increase the distance between the outer edge of the solar panels and the disturbance wake from the shield during the aerobrake stage. The magnetometer is deployed in the final orbit and is extended 5.25 meters away from the bus. This extension is accomplished by three actuating joints.

MATERIALS

The materials chosen for the E-MTV shells and plates were the result of analysis performed on several possible choices. The analysis rated the materials in their ability to meet several different needs in this design.

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	Low Cost	Machinablility	Low Density	Thermal Conduction Capabilities	Longitudinal Strength
Aluminum Alloy	5	5	4	5	4
Graphite Epoxy	2	4	5	0	5
Titanium	1	4	2	2	5
Boron/ Aluminum	4	5	4	2	5

This analysis led to the selection of aluminum alloy as the main material for the E-MTV. One of aluminums best characteristics is its ability to conduct heat to the colder parts of the E-MTV. This will be discussed in the thermal control section.

To calculate the thickness needed in the shells, a force analysis was done using an equation relating critical buckling stress of shells to its thickness. For the relay bus, a calculated load of 168,000 to 360,000 newtons of force may act through its shell during the initial acceleration. Using the equation:

$$\text{critical load} = 1.2(\text{pi})Et^2$$

a value of .1 cm is found to be the thickness at which the relay bus would buckle. Using a thickness of .3 cm would assure a stable shell during accelerations. Similar analysis were done on the conical shell using the equation:

$$\text{critical load} = .3994(\text{pi})Et^2(\cos \text{alpha})^2$$

These equations determined the thickness of the shells needed and the corresponding masses. A thickness of .3 cm was used around the E-MTV except for the conical shell which needed a thickess of .4 cm.

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The materials needed for the aerobrake shield must be able to withstand high temperatures. Carbon/carbon composites can easily provide protection up to temperatures of 2000 K. Carbon/carbon (cc) composites were chosen over ceramic composites because of their 50 percent decrease in density. The problem of oxidation of the cc would be minimal during this mission.

THERMAL CONTROL

With 1300 W/M^2 of solar energy reaching the E-MTV at Earth, and 600 W/m^2 at Mars, the E-MTV will have reflective multilayer insulation to reduce the amount of energy absorbed into the aluminum shells. The aluminum shells will conduct some of the energy to the colder side. To maximize the amount of heat transfer between the warm and cold side, a highly emissive black paint will be coated on the inside of the shells. This paint will take heat from the warm shell and radiate it across the inside of the shell. This passive control will satisfy most thermal requirements. The components which require rapid heat dissipation are equipped with heat pipes. The thermal control of the liquid fuel during the trip to Mars will be accomplished by active components figured in with the fuel tanks.

COMPATIBILITY AND ASSEMBLY

The space station remote manipulation device (rmd) will be able to control the constructed E-MTV by a grapple place on the shell of the fuel tanks. This grapple is placed directly on the outside of the E-MTV at the center of mass. This location is shown in figure 2.2. The same grapple will be used by the rmd on the space shuttle. The components of the mission will be carried to the space station by the space shuttle in the manner shown in figure 2.5. The trunions shown on the shell housing the fuel tanks can be adapted to fit into

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the shuttle cargo bay using attachment numbers 166,198 and 225. The bus will be supported by the fuel tank section underneath it.

The remaining cargo capsules will be constructed to the dimensions shown in figure 2.5. These dimensions are the minimum needed to transport the remaining cargo to the space station.

Assembly of the E-MTV will require plenty of area aboard the space station. The E-MTV should be constructed from the rear to the front of the craft with the shield being formed and fitted last.

Instrument bus	Payload Capsule	Fuel Tanks/Support
Stablilizing platform	Front airfoil pieces	2 - H_2 tanks
Science plate	Rear airfoil pieces	2 - O_2 tanks
Extending Rods	Fuselage	2 - Thruster packs
AAC electronics	Crate #1 (Electronics, solar cells, lasers)	Support structures
Rear support plate	Crate #2 (Mechanical parts)	
2 - Communication	2 - Rear fins	
2 - Solar arrays	Cylindrical shell	
2 - Extendable masts	Front and Rear plate	
4 - Fuel tanks	Conical shell	
1 - Magnetometer	Parachute canister	
4 - Thruster packs	2 - Gyroscopes	
10 - Science components	2 - Fuel tanks	
1 - Momentum wheel	2 - Batteries	
1 - Gyroscope pack	Landing gear	
6 - Accelerometers	Aerobrake shield	
1 - Computer	4 - Thruster packs	
1 - Tape Drive		
1 - Multiplexer		
2 - Sun sensors		
2 - Solar Array Actuators		
2 - Star trackers		
1 - Thrust cone		
1 - Low gain radio		

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SPACECRAFT CONFIGURATION AT LAUNCH

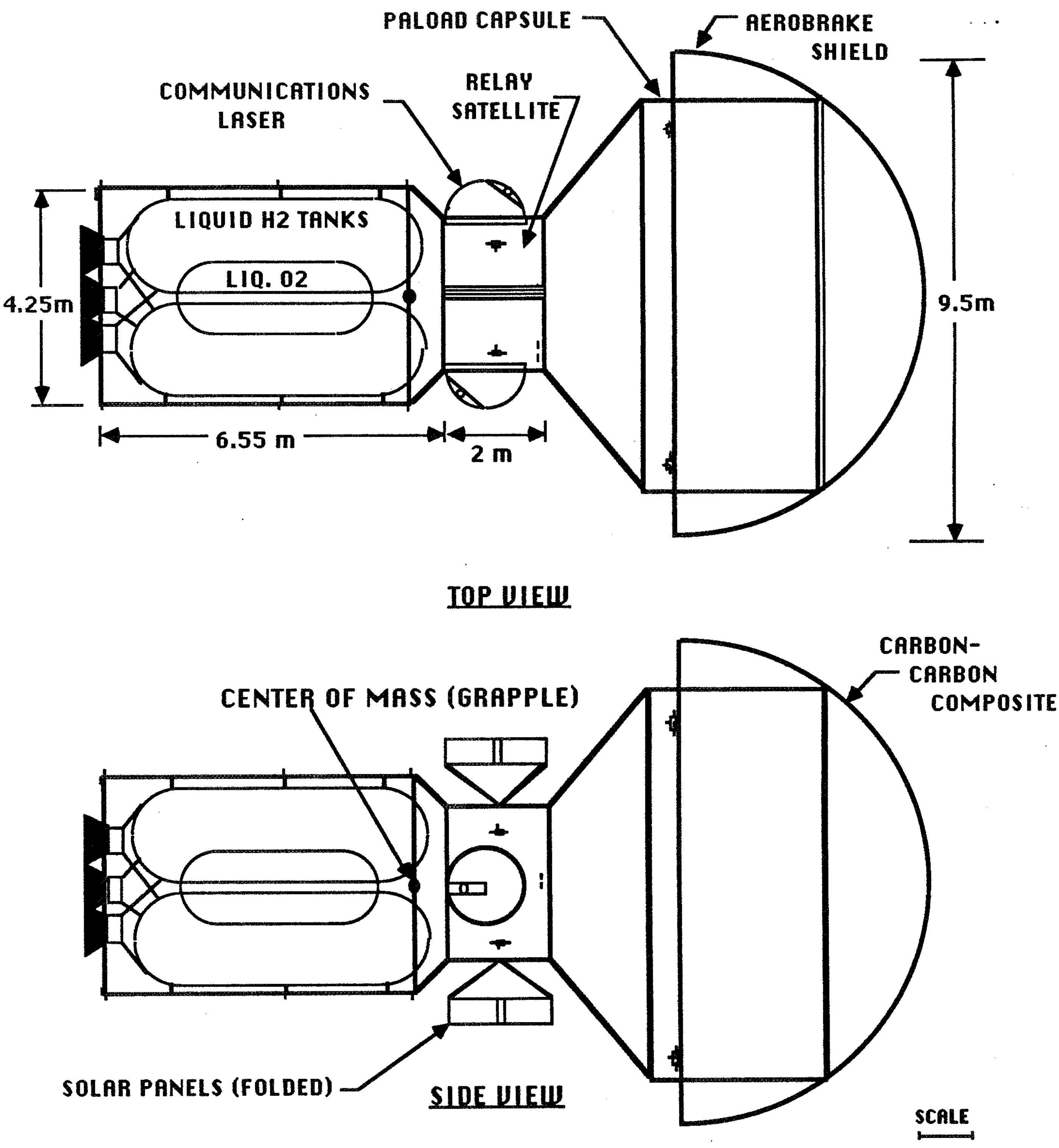

figure 2.1

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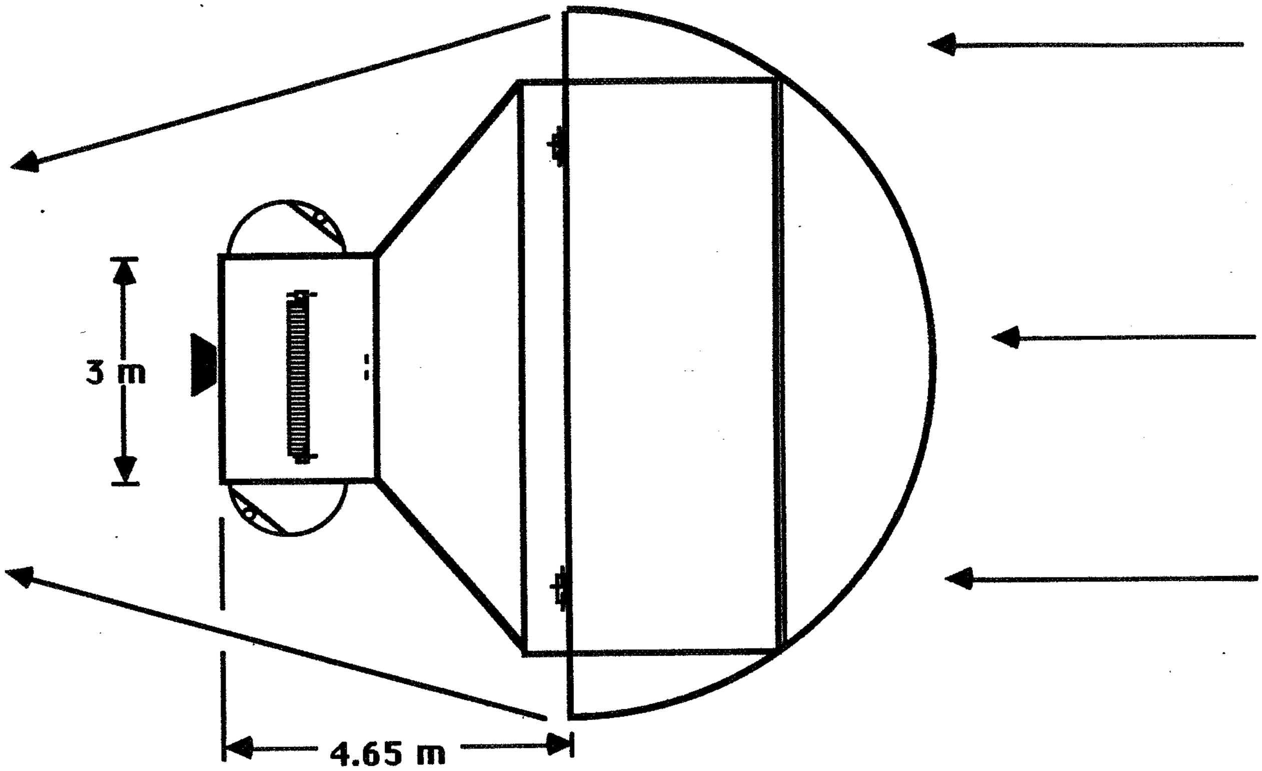
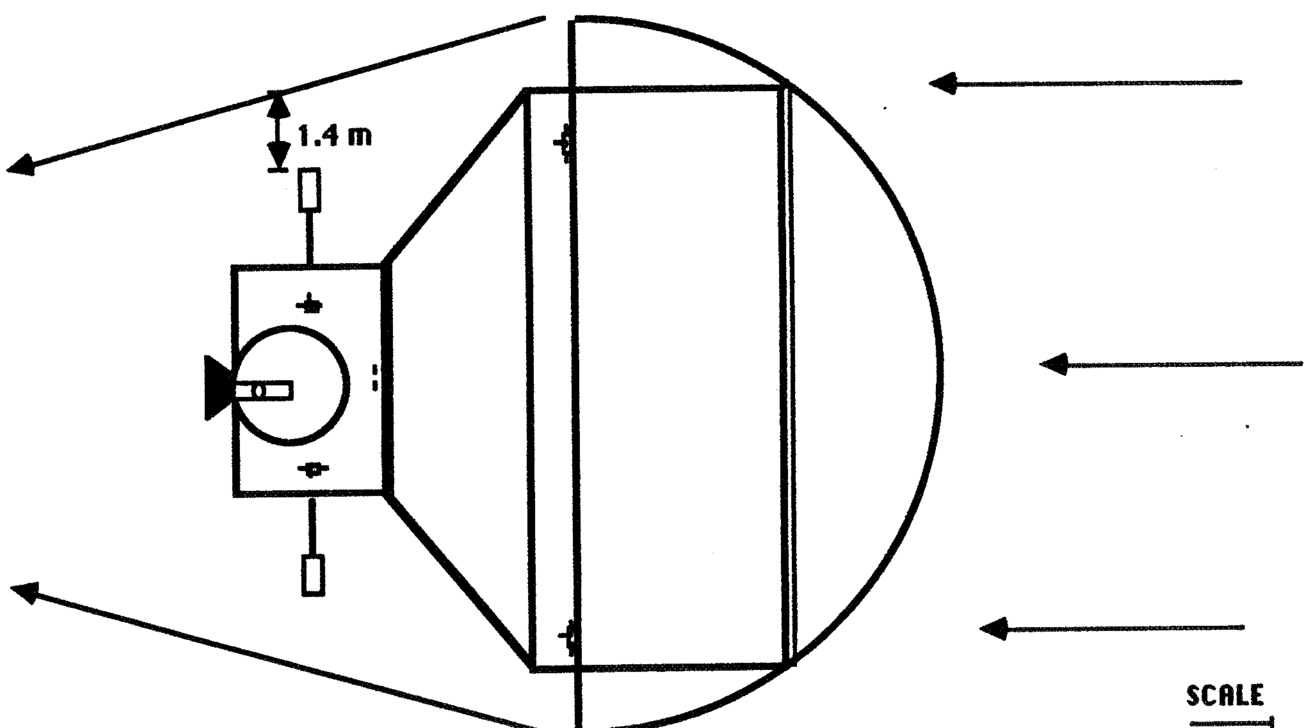
SPACECRAFT CONFIGURATION DURING AEROBRAKING



TOP VIEW



SIDE VIEW

figure 2.2

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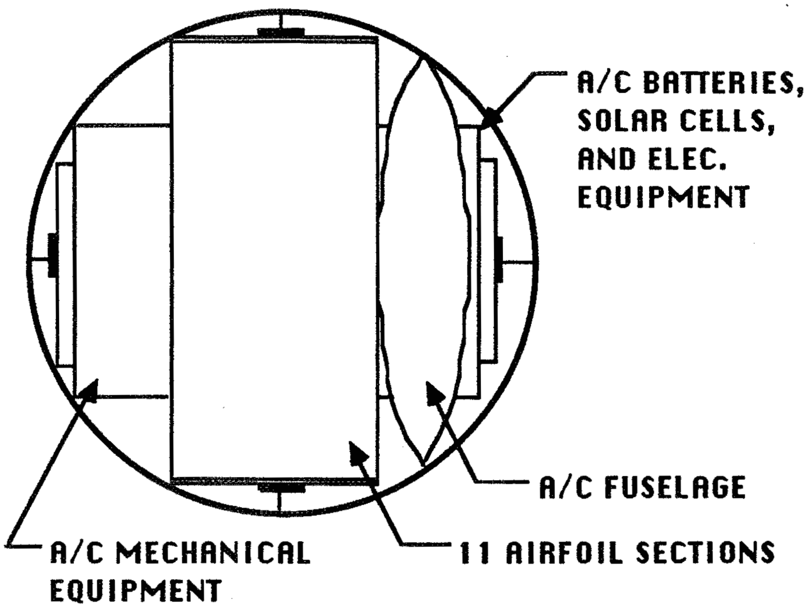
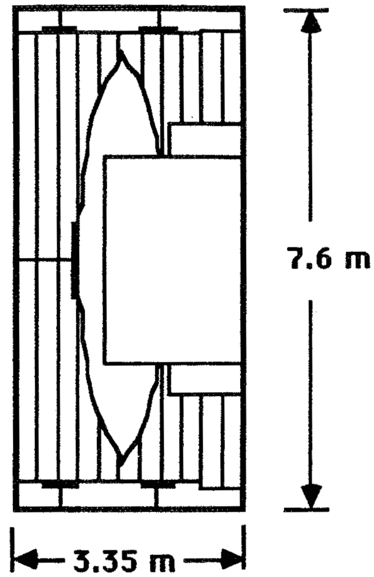
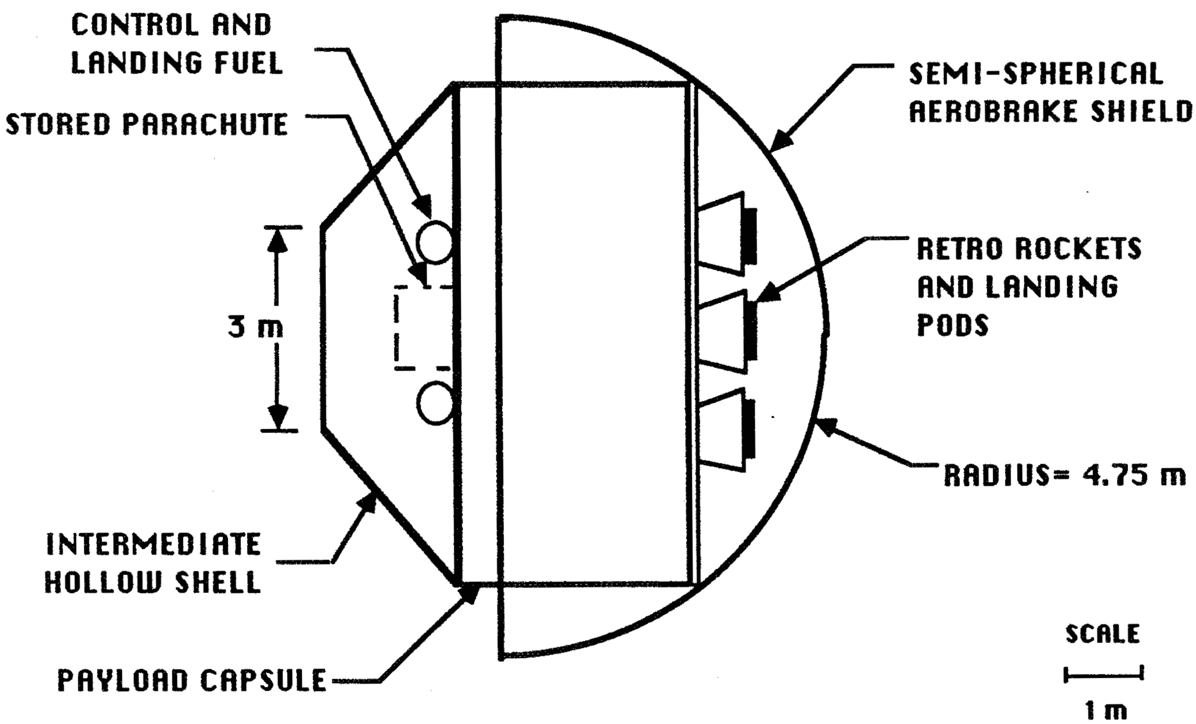
PAYLOAD CAPSULE AND AIRCRAFT CONFIGURATION



AXIAL VIEW



SIDE VIEW



PAYLOAD CAPSULE IN SHIELD

figure 2.3

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RELAY SATELLITE

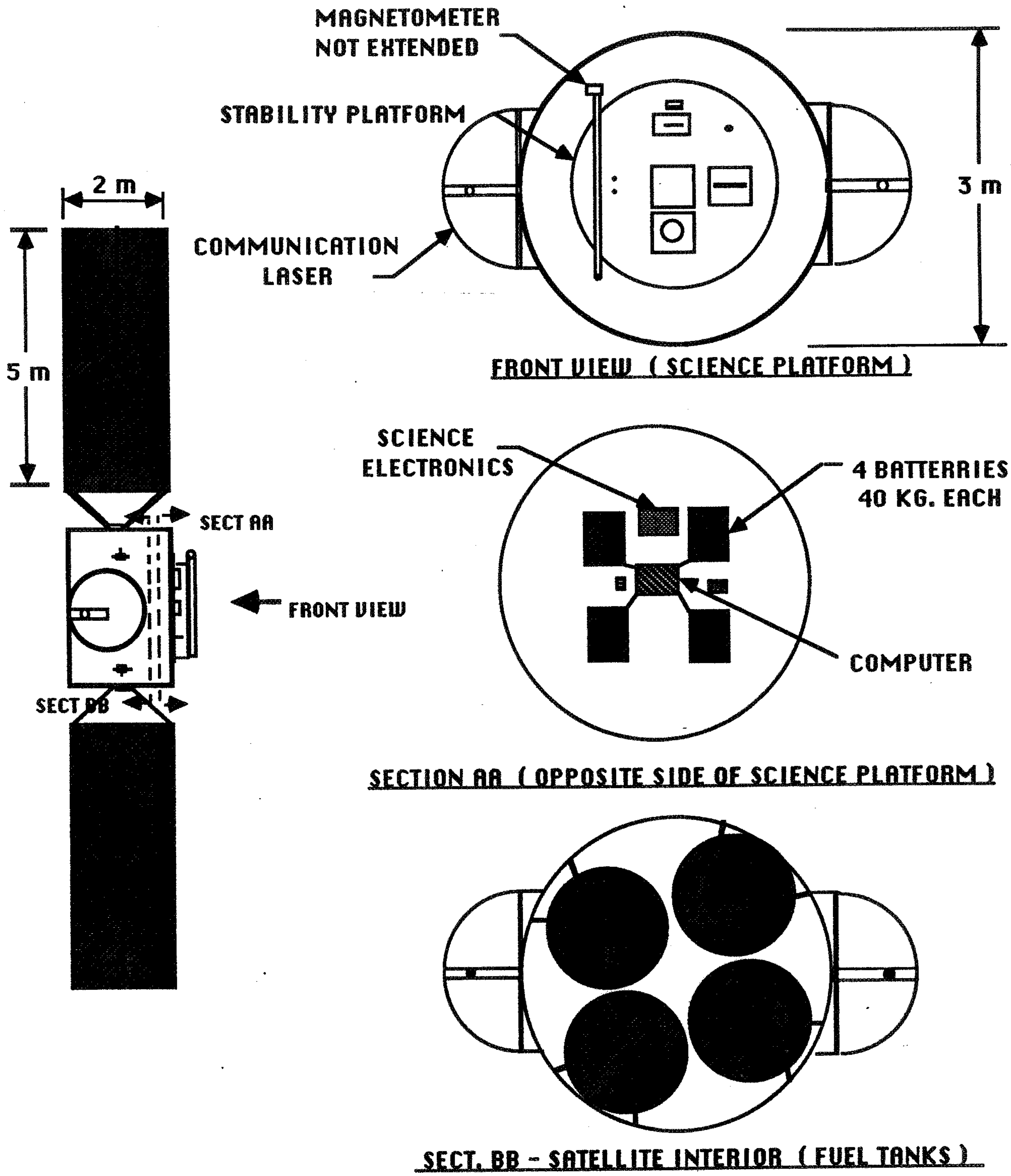

figure 2.4

SCALE
1 m

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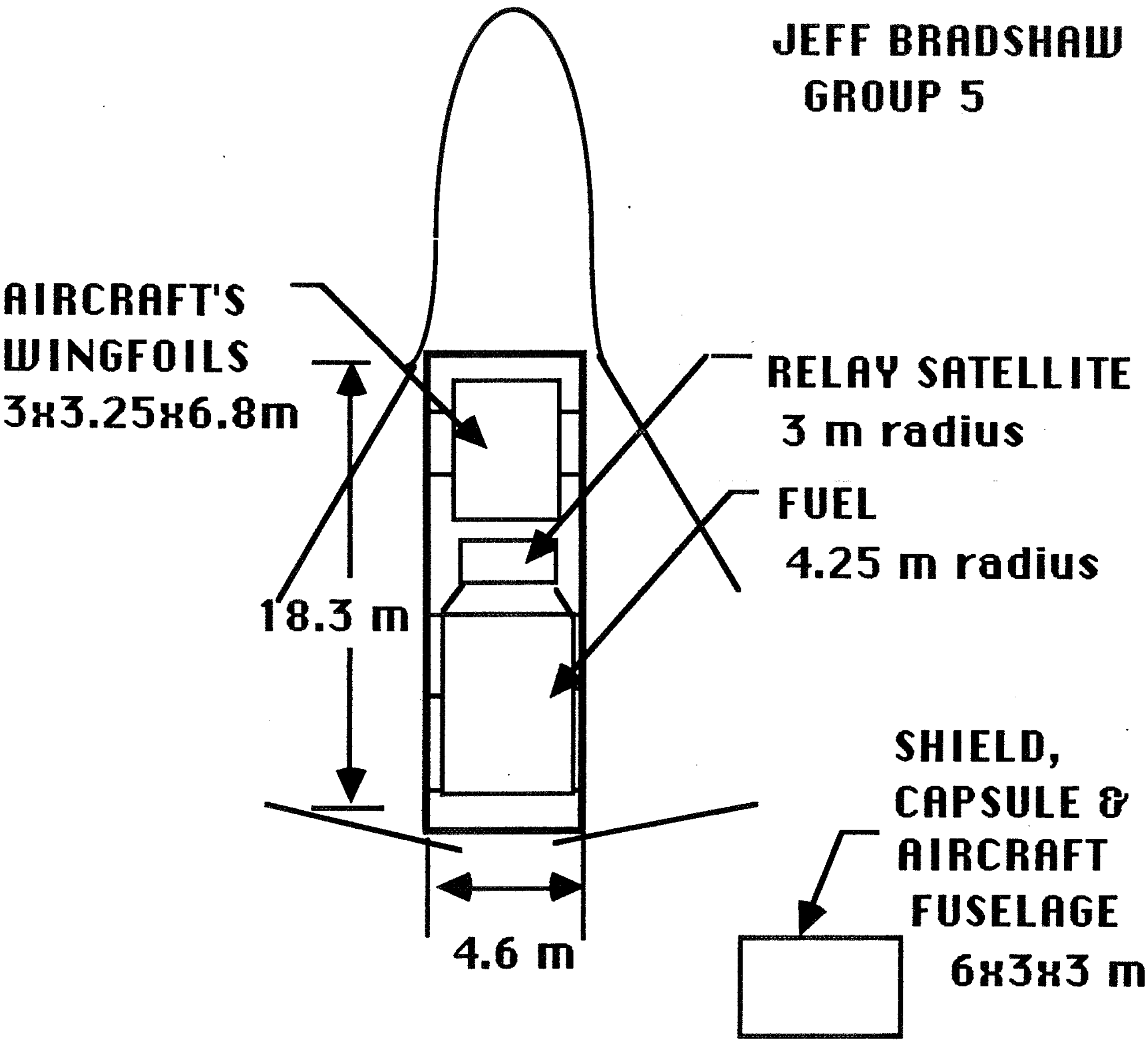

1 1/2 SHUTTLE TRIPS REQUIRED

SHUTTLE COMPATIBILITY

FIGURE 2.5

- **Components**

  The C^3 will require three components as per the RFP.

  These three components are:

  - **Computer**
  - **Radio**
  - **Antenna**

SELECTION OF THE RADIO/ANTENNA:

The requirements pertaining to the selection of the radio/antenna are listed as follows in order of importance:

1. Stress simplicity, reliability, minimum mass and low cost.
2. Should not use materials or techniques expected to be available after 1998.
3. Should use off-the-shelf hardware where available.
4. Should have a minimum design lifetime of four years.

Two different radio/antenna packages were looked at for the E-MTV. They were the radio/antenna package used on Voyager and an optical radio/antenna called OPTRANSPAC (Optical Transceiver Package). The following list is a comparision between the two packages.

COMPARISION BETWEEN VOYAGER ANTENNA AND OPTRANSPAC FOR USE ON THE E-MTV

	Voyager Antenna	OPTRANSPAC
Dimensions (m)	3.7 diameter parabolic dish	1.07 x 0.46 x 0.61
Frequency	X-band 8450 MHz	Visible laser 5.64×10^{14} Hz
Uplink Rate (kbps)	10	100
Downlink Rate (kbps)	na	1
Antenna Gain (dB)	48.0	155.42
Transmission Efficiency	62%	70%
Space Loss (dB)	-301.0	-390.96
Power (Watts)	20	57
Weight (kg)	na	52.2

Both of these packages satisfied the RFP requirements. Even though the Voyager radio/antenna has already been space tested and has the lesser of the two power requirements, the OPTRANSPAC was chosen because of the much higher transmission rate and the reduced size. The reduced size has the advantage in that it is much less bulky and easier to guide. Also shielding a parabolic dish from the high temperature encountered during aerobrake and high velocity space particles would be more difficult. Another factor in chosing the OPTRANSPAC was the fact that its beam divergence was much

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Blankenship, C; Teichman, L; Advanced Materials Technology, National Aeronautics and Space Administration Publication 2251, 1982.

Agrawal, B; Design of Geosynchronous Spacecraft, Prentice-Hall Inc., 1986.

Allen, D; Haisler, W; Aerospace Structural Analysis, John Wiley & Sons, Inc., 1985

POWER AND WEIGHT BREAKDOWN OF THE OPTRANSPAC FOR USE ON THE E-MTV

COMPONENT	WEIGHT (KILOGRAMS)	POWER (WATTS)
Electro-Optics Assembly	[25.86]	(11.0)
Telescope Assembly	8.16	-
Imaging Optics Assembly	(6.12)	2.0
Optics Baseplate	2.01	
Dust Covers	1.04	
Light Tube	0.13	
Optics Components	2.49	
Wire/Connectors/Fasteners	0.45	
Laser Assembly	(9.07)	4.0
Structure	5.71	
Optical Bench Assembly	2.27	
Diode Array Assembly	1.09	
Detector Assembly	0.45	0.4
Earth Tracker	(2.04)	4.6
Head Assembly	1.36	
Electronics	0.68	
Electronics	[18.60]	[39.3]
Communications Electronics	(4.08)	7.7
Circuit Boards	2.37	
Mother Board	0.20	
Wire/Connectors	0.61	
Structure	0.90	
Control Electronics	(5.67)	14.5
Circuit Boards	3.31	
Mother Board	0.31	
Wire/Connectors	0.83	
Structure	1.22	
Power Conditioner Assembly	(8.85)	17.1
Modules	5.44	
Wire/Connectors	0.59	
Structure	2.81	
Structure/Wire/Miscellaneous	[7.71]	6.7
Wire/Connectors	4.76	
Support Structure	1.18	
Miscellanous Structure	1.77	
Total	51.17	57

Two OPTRANSPAC's will be mounted on the E-MTV. They will be placed diametrically opposite to each other at ninety degree angles from the solar panels as shown in Fig 2.

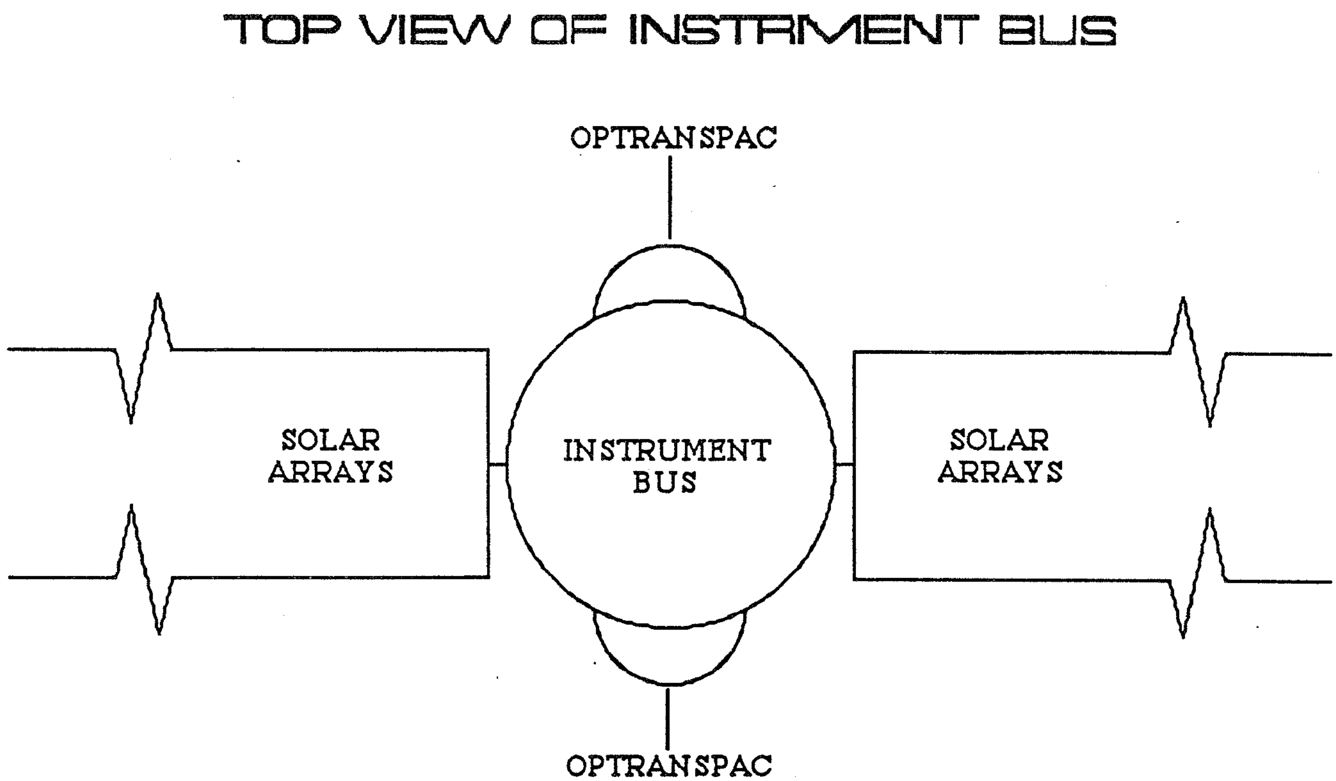

Figure 2

In addition to the redundency, they allow an almost full sperical field of view for communication purposes. If one of the units should fail, the other will be able to takeover with a slight bit of spacecraft manuevering. The only restriction of the field of view of the OPTRANSPAC's is the aerobrake shield in the front and the fuel tanks in the rear. In Fig 3, you can see one of the OPTRANSPAC's mounted on the instrument bus. The aerobrake shield causes a 37 degree "blank spot" on the aerobrake shield side of the E-MTV and a 24 degree "blank spot" towards the rear of the E-MTV. This will not be a problem for the majority of the mission.

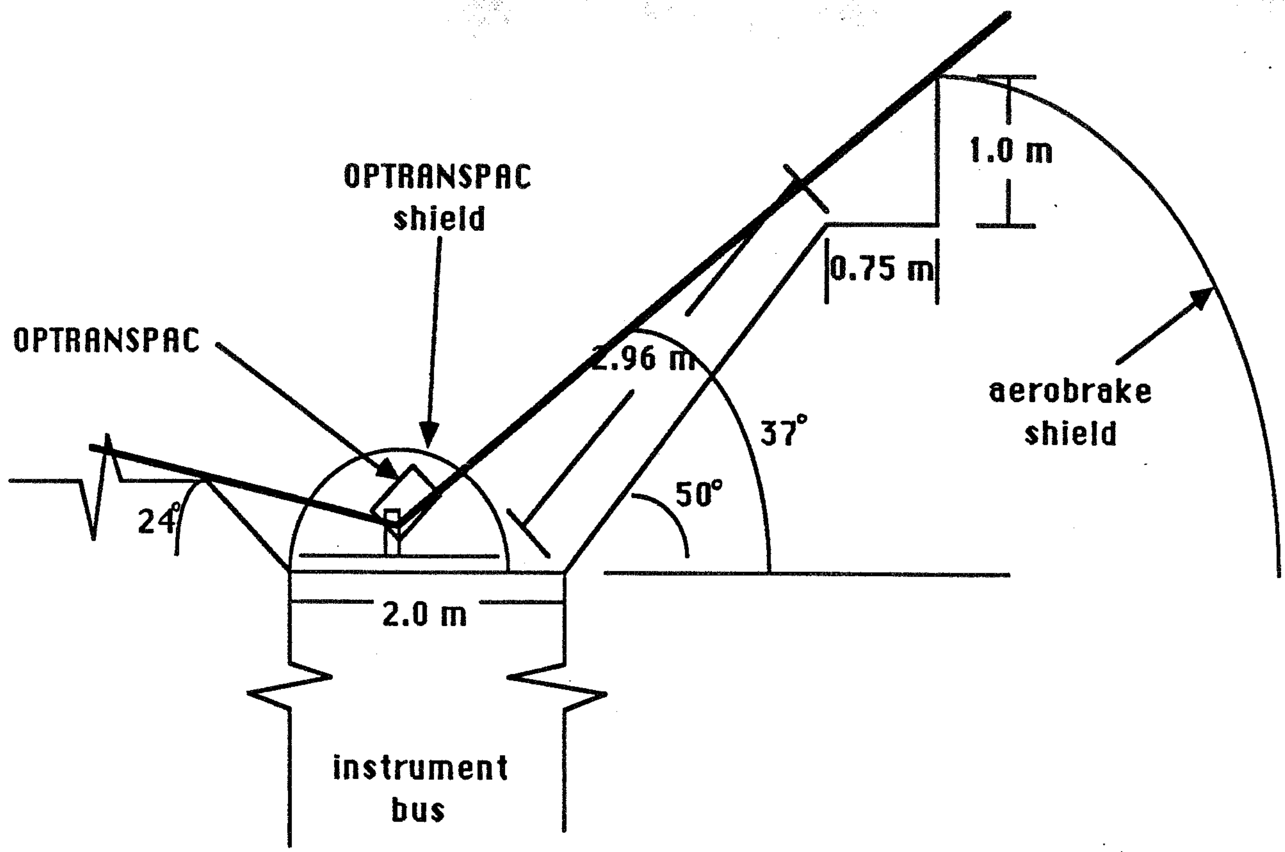

Figure 3

From Mars, there will be an 8.64 microradian divergence of the beam. This will be sufficient to ensure accuracy of pointing the beam towards Earth. Fig 4 shows the E-MTV orientation for the OPTRANSPAC pointing during Earth departure and Mars arrival.

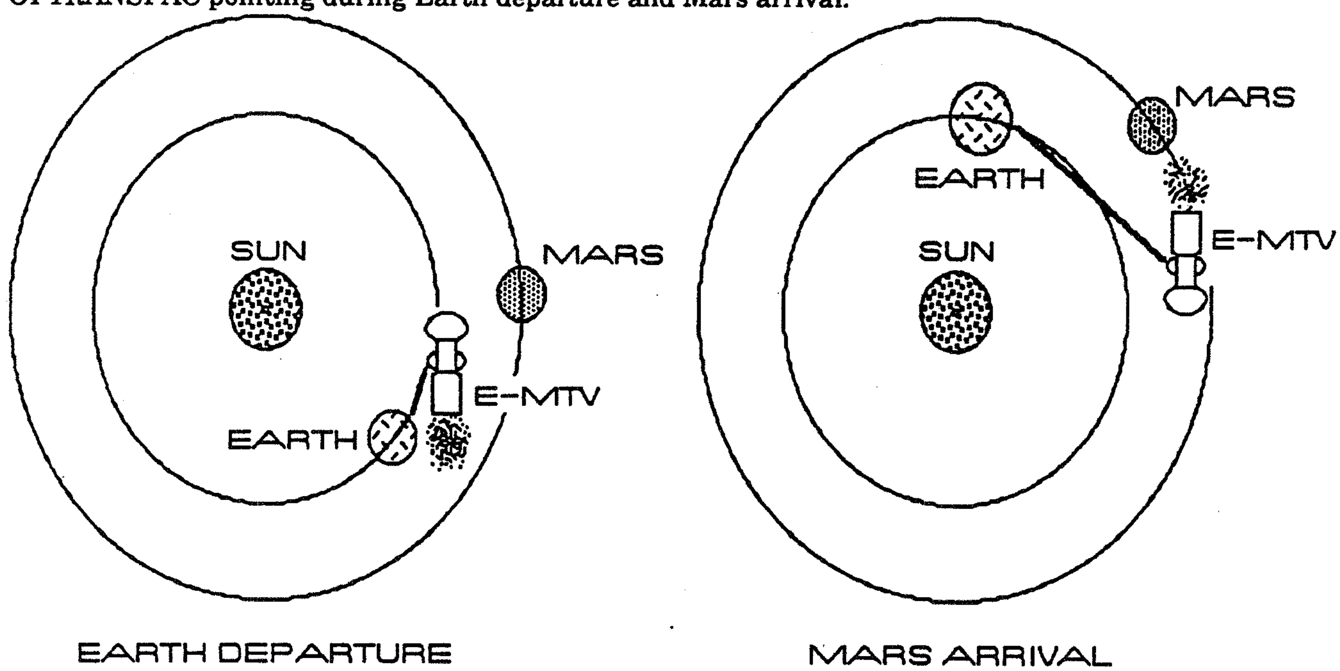

Figure 4

As we leave Earth, one of the OPTRANSPAC's will be pointing towards Earth. Somewhere en route to Mars, the Earth will actually pass Mars causing a temporary communications blackout since the shield prevents any transmission directly ahead of the spacecraft. As we approach Mars, however, the

E-MTV will be rotated 180 degrees in order to fire the engines to achieve an elliptical Martian orbit. During aerobrake, there will be another communications blackout due to high temperatures and perhaps ionization of the atmosphere surrounding the E-MTV. After aerobrake, the E-MTV will be in a low Mars orbit of about 350 nautical miles above the surface for mapping purposes and the viability of the predetermined landing site for the payload. At this point, the payload will separate from the instrument bus. However, it will remain in orbit only a short distance from the instrument bus until it reaches a signal from the instrument bus via a low gain radio to commence reentry procedures. During this time, the E-MTV will be orientated as shown in Fig 5. In this position, one of the OPTRANSPAC's could be relaying data to the Mars base while the other will be relaying the same data to Earth. This will be an on-and-off process as both the Mars base and Earth will be out of view for long periods of time during each orbit.

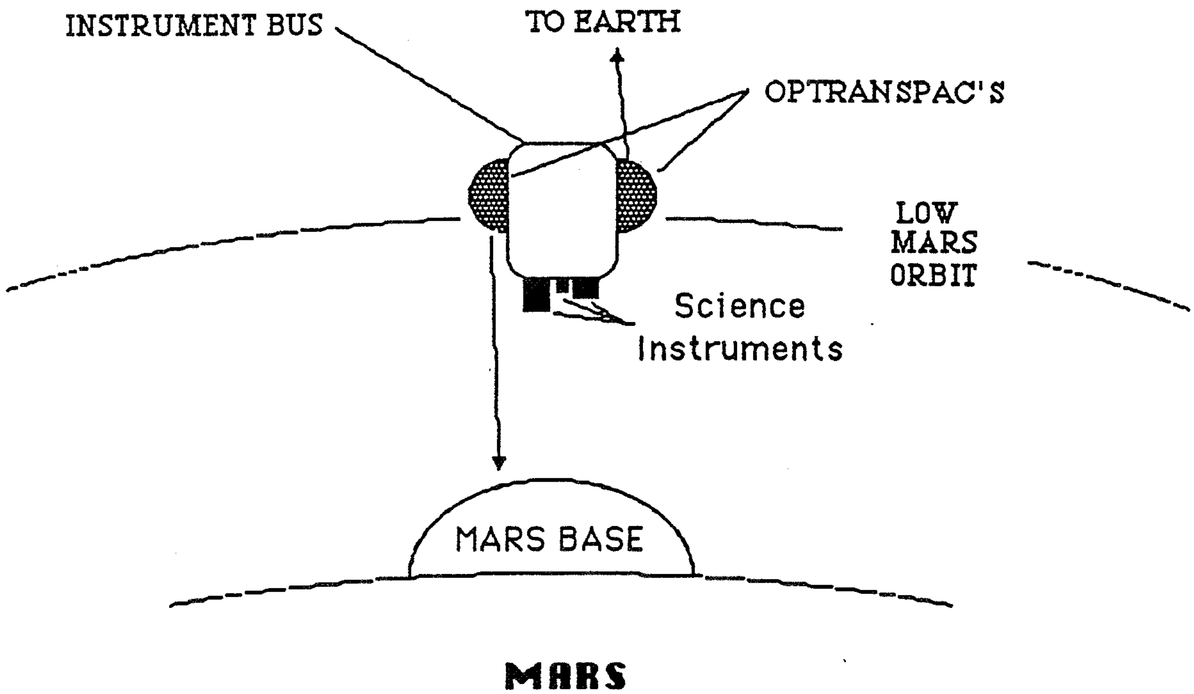

Figure 5

After receiving the go-ahead from the Mars base, the instrument bus will send a signal to the payload to commence reentry. At this point another burn will carry the E-MTV to a stationary Martian orbit above the base. After the aircraft has been assembled, the instrument bus will then commence communications with the aircraft. The orientation will be similiar to that in Fig. 5 except that the instrument bus will be in a synchronous Mars orbit and it will be communicating with the Earth and the aircraft. Communications with Earth will be not be able to take place each time the E-MTV is situated such that Mars is between it and Earth. Every time this happens, all data to be transmitted to Earth is stored in a data storage device until communications are possible once more. This will be controlled by the onboard computer explained in the next section. This concludes the Radio/Antenna implementation phase.

SELECTION OF THE COMPUTER:

The selection of the computer was based on several different criteria. They are listed in order of importance:

1. System Capability

   Memory: This included both the amount of memory physically installable in the computer and the amount of memory addressable by the central processing unit.

   Input/Output: This consisted of whether the computer had DMA (Direct Memory Access), and if so, what was the DMA rate. It also included the amount of input/output terminals and the type (analog/digital). Lastly, this category included computer speed in kilo-instructions executed per second (KIPS).

2. Radiation Tolerance: There are several types of radiation hazards faced by spaceborne components, but the one of primary importance used is total dose capability. For most space missions, 10^4 Rads (Si) is often considered a minimal total dose tolerance. Another type of radiation hazard experienced by spaceborne computers is the single-event upset (SEU). This is usually expressed as errors/bit/day for a particular orbit or position in space. The SEU occurs when a high energy atomic particle causes a bit in the microprocessor or memory chip to flip. The amount of damage caused could be hardly noticable (e.g. in a data RAM chip) at best to catastrophic (e.g. in an important line of code in the CPU logic) at worst.

3. Weight and Size: This had a direct effect on the amount of fuel used and the placing of the computer in the instrument bus. Therefore, minimizing both weight and size was desirable.

4. Power Consumption: This is a difficult category to compare since the power consumption in a computer usually depend on the size of memory and the microprocessors used. For this particular comparision, above 100 watts was cause for rejection. The following is a list of several computer packages which met the specific requirements for this mission.

COMPARISION OF COMPUTER PACKAGES

FOR USE ON THE E-MTV

Model	CPU speed (KIPS)	RAM access (nanoseconds)	Size/type of RAM chips
DELCO M372	590	180	16K SANDIA
DELCO M572 Magic V	700	180	16K RCA
GENERAL ELECTRIC	550	300	16K RCA
IBM 1750A AVIONICS	1000	70	na
Litton ATAC AVIONICS	500	180	16K RCA
RCA SCP	530	170	16K RCA
TELEDYNE 750	550	150	16K SANDIA
TRACOR RH 1750A	400	200	16K SANDIA
McDonnell Douglas	2000	na	256K GaA chips

Model	Power (Watts)	Weight (kg)	Size (cm^3)
DELCO M372	126	23.6	24352
DELCO M572 Magic V	55	12.7	13026
GENERAL ELECTRIC	80	8.2	8495
IBM 1750A AVIONICS	95	9.4	7646
Litton ATAC AVIONICS	112	17.2	14725
RCA SCP	26	15.0	24919
TELEDYNE 750	197	35.4	31715
TRACOR RH 1750A	39	6.4	21238
McDonnell Douglas	na	25	4719

The McDonnell Douglas GaA 32-bit vector processor was a latecomer in the selction. However, it ended up being chosen for this particular application for many reasons. Among them were its superior speed and smaller size. Another reason was because this particular computer package is extremely radiation resistance. It was resistant to radiation amounts of up to 10^8 rads. This is well above the minimum required for this mission. It uses eight 32-bit processing elements connected in parallel for an effective clock rate of over 200-MHz. These processors are all on separate modules making this multifunction computer expandable. All this is packed into a box about the size of a coffee can. The computer itself has enough random access memory chips for short time storage. However, when we start receiving data from the science intrumentation, especially the atmospheric sounder, there will be far too much data to transmit instantly or to store in computer memory. A costly solution would be to increase the number of RAM chips. However, for the number of chips required, it would complicate the circuitry as well as increase power consumed by the computer system. Another solution would be to add a magnetic tape drive. This would provide a buffer to store data until it is ready to be transmitted. A tape drive admittedly has much more mass than RAM chips but the advantages outweigh this slight deficiency. It has a much lower cost to data storage ratio than computer chips. Also, if we wanted to increase the memory for a system utilizing RAM chips we would have to integrate more chips with the rest of the circuitry. However, if we wanted to do the same with a tape drive, the solution would be to simply increase the tape length. It is apparent that the tape drive is the superior component in terms of cost efficiency and flexibility. A possible tape drive to be used for this mission would be a multitrack tape recorder. Multitrack tape recorders have the advantage in that they can read/write on several longitudinal tracks at one time. The data rate on some multitrack recorders range from 60 to 200 Mbps. The capacity of such a recorder can approach 250×10^9 bits.

There will be many inputs to the computer from the various subsystems. These inputs will be coming mainly from the attitude and articulation subsystem and the science subsystem. Some of these inputs will be in analog form and will have to pass through an analog-to-digital (A/D) converter before they can be sent to the computer. Some of the data rates coming in are extremely high. There will not be enough onboard memory or even tape capacity to store all this information for long periods of time. Therefore, some of this data will have to be compressed. This can be achieved by passing the inputs through some sort of a multiplexer. One of the more promising multiplexers being looked at for Project ACRONYM is the Multi-megabit Operational Multiplexer System (MOMS). This multiplexer has 60 channels and can receive data at up to 280 Mbps. This will sufficiently take care of the data from the science instruments. The MOMS also has the capability of compressing data at a 7 to 1 ratio. This means that it is possible to take in 7 bits of data and compress it into one. This will only be done to the Atmospheric Sounder and the Radar Altimeter Mapper. There will be a loss in resolution, but not sufficient to hamper the mission. The following table shows different types of input and the data rates from some components.

INPUT TYPES AND RATES OF SEVERAL COMPONENTS

INTERACTING WITH E-MTV COMPUTER

SUBSYSTEM	COMPONENT	FORM OF INPUT (A/D)	DATA RATE (IF APPLICABLE)
AACS	2 - Narrow Angle Sun Sensor	4 analog inputs per sensor	na
AACS	2 - Wide Angle Sun Sensor	4 analog inputs per sensor	na
AACS	3 - Gyroscope	3 analog inputs per gyroscope	na
AACS	3 - Accelerometer	1 analog input per accelerometer	na
CCC	OPTRANSPAC	digital	1 kbps
SCI	Gamma Ray Spectrometer	digital	1.3 kbps
SCI	UV-VIS-IR Reflectance Spectral Mapper	digital	3 kbps
SCI	Magnetometer	digital	200 bps
SCI	Radar Altimeter Mapper (RAM)	digital	30 kbps
SCI	UV Photometer	digital	500 kbps
SCI	Radio Science	digital	1 kbps
SCI	Atmospheric Sounder	digital	2 Mbps

Some of the outputs/commands the computer will give will be the telemetry and commands to the OPTRANSPAC to be transmitted, the commands for the solar array pointing, the engine firing sequences, and to the payload section prior to separation to name a few. The power switching is to be controlled via a power distribution unit (PDU) which is part of the power and propulsion subsystem. For more information on the PDU, please refer to the power and propulsion subsystem section of this report. The computer and its various components will be mounted in the interior of the instrument bus as shown in Fig. 6.

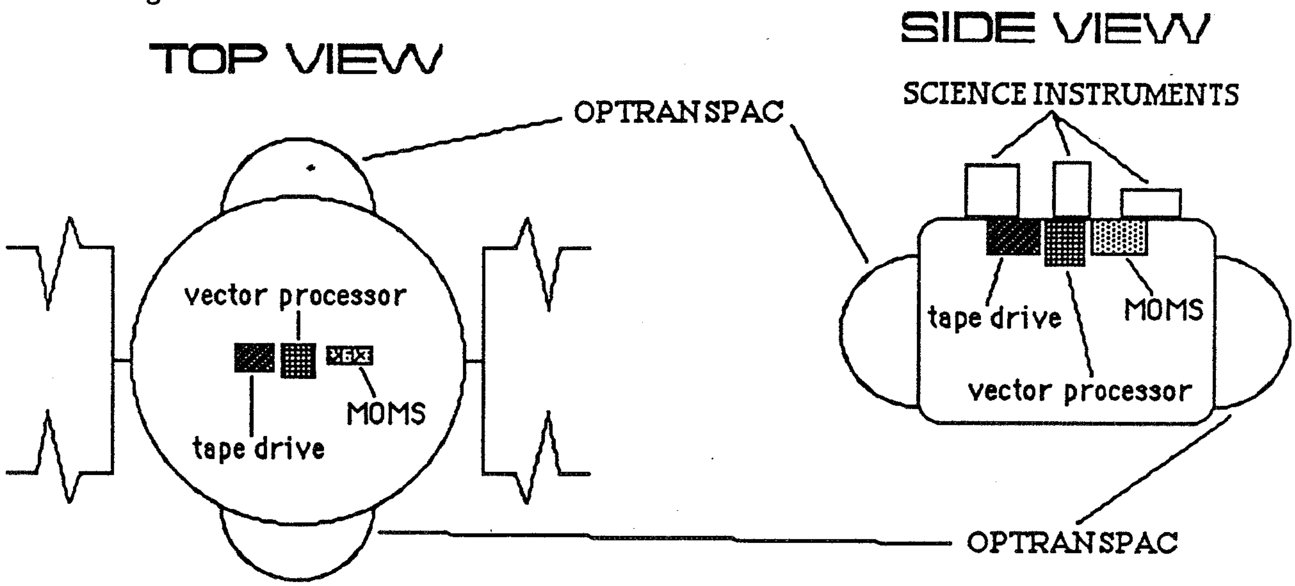

Figure 6

TECHNICAL PROBLEM AREAS

There are a few technical problems still remaining in both the communications and the computer subsytems. For the communications, excessive spacecraft "jitter" might render the OPTRANSPAC inoperable for various amounts of time since it will not be able to maintain a communications link with the receiving station. Some method of vibration damping system will have to be utilized in order to minimize this as much as possible. Interfacing the sun sensors with the OPTRANSPAC might also be a problem. The additional weight could cause, for example, a different damping ratio to be used for the telescopes' equations of motion in the target pointing program in the OPTRANSPAC. Therefore, the OPTRANSPAC would have to be modified in both hardware and software. In the area of the computer/components, thermal control might pose a problem, especially during aerobrake. Passive thermal control might not be sufficient to handle the heat produced in which case, some method of active thermal control will have to be utilized. This would, however, cause a dramatic increase in both mass and power consumption.

SUMMARY

After reviewing many component packages which sufficiently met the mission requirements, the OPTRANSPAC and the McDonnell Douglas 32-bit vector processor were chosen as the communications and computer package, respectively. The Multi-megabit Operational Multiplexer System along with a high density multitrack tape drive were tentatively chosen for input conversion, data reduction and data storage. Two OPTRANSPAC's are to be mounted opposite each other on the instrument bus to ensure maximum communication coverage. They will both be shielded by a hemispherical cover of 1 gm/cc aluminum. The computer and components are to be mounted on the inside of the instrument bus just under the science instruments. During the mission, the only commands and inputs the C^3 will be receiving will be from the AACS and the PPS. The science subsystem will not be active until after payload separation. After the payload has received the signal to commence reentry the E-MTV will move to a stationary Mars orbit and will begin sending telemetry back from the science instruments. After this stage has been completed, the instrument bus sill continue to provide support to the operation of the aircraft.

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Materials Research Lab

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EARTH- MARS TRANSPORT VEHICLE

SCIENCE SUBSYSTEM

TOBY B. MARTIN
DESIGN GROUP #5

MAY 3, 1988

Science Objectives

The Request for Proposal (RFP) outlines the development of a Mars aircraft space delivery system. The Earth-Mars Transport Vehicle (EMTV) will consist of a payload capsule and an instrument bus. The payload capsule will by design descend to the planet surface, while the instrument bus remains in orbit to perform scientific experiments and act as a relay satellite supporting the aircraft. The RFP also gives project guidelines and general requirements for the development of the system. A list of the requirements given in the RFP that concern the Science Subsystem (SCIS) is found in Table (SCI-1).

RFP REQUIREMENTS
1. Spacecraft will consist of two components a) payload reentry system b) orbiting instrument bus
2. A number of subsystems are defined for integration, one of which is the Science Subsystem
3. Nothing in the design should preclude it from performing several possible missions
4. Design lifetime of 4+ years
5. Stress simplicity, reliability and low cost
6. Fulfill objectives outlined in 'Mission Science Objectives' document

Table SCI-1

SCIENCE INSTRUMENTATION REQUIREMENTS
* Meet Mission Science Objectives
*Collect Data
*Send Data to Command, Control, and Communications Subsystem (C^3)
*Receive power from Power and Propulsion Subsystem
*Receive Commands from C^3
*Select Components
*Minimize Cost

Table SCI-2

Specific derived requirements that must be fulfilled are given in the list 'Science Instrumentation Requirements' (Table SCI-2). Finally the list of 'Mission Science Objectives' (Table SCI-3) is given to round out the ISS requirements.

The RFP has many ambiguities within its structure. some of the main ambiguities that concern the SCIS and the clarifying assumptions made are given in the next few paragraphs.

The main ambiguity found in the RFP is the hierarchy of roles the spacecraft is to play. Which is more important - payload transport, relay satellite or the role of scientific platform? A discussion between all systems analysts working on the project came up with the assumption: First, and foremost the spacecraft is a payload transport, the next priority level went to the relay satellite to support the aircraft and base station in their operation, and finally the satellite is a science platform. The rational being that the development of the spacecraft was devised to carry the Mars aircraft to the planet. Along with the assumption that the planet must have been thoroughly explored from space before building the base station. This tertiary role for the science group brings up requirement conflicts during both development and mission sequences. Each conflict will be noted and addressed throughout the text.

MISSION SCIENCE OBJECTIVES

General Objectives:

1. Determine the origin, evolution and present state of the Solar System
2. To better understand the Earth through comparative planetary studies
3. To understand the relationship between the chemical and physical evolution of the solar system and the appearance of life.

Specific Objectives:

1. Determine the elemental and mineralogical character of the Martian surface on a global basis
2. Determine the distribution, abundance, and concentrations of volatile materials and dust
3. Define the global gravitational field
4. Measure the global topography
5. Explore the atmospheric structure and its circulation in detail
6. Establish the nature of the Martian magnetic field

Table SCI-3

Component Selection

With the RFP and the SCIS requirements in hand a methodological approach to meeting each requirement was developed. A general design approach (Fig. SCI-1) was modified specifically for the SCIS component selection process (Fig. SCI-2). These general approaches were followed during each component selection.

Define Requirements

Select S/C Components

Integrate Components

Test and Simulation

Final Design

Fig. SCI-1

Define Requirement
Literature Search
Instrument List
Meet Requirement?
List of suitable Instruments
Feature Comparison
*weight, size, power
*cost
*reliability
*performance
Satisfactory?
Selected Component

Fig. SCI-2

Because of space limitations only one specific instrument selection will be demonstrated. Each of the other instruments selected came under similar scrutiny.

*Define Requirement
-Measure the global topography

*Literature search resulted in these suitable instruments:

-Synthetic Aperture Radar (SAR)	(Hord, p. 45)
-Thematic Mapper (TM)	(Hord, p. 64)
-Radar Altimeter Mapper (RAM)	(Fimmel, p. 58)
-Return Beam Vidicon Camera (RBV)	(Hord, p. 99)
-Charge Coupled Device Imager (CCDI)	(NASA SP-335, p. 275)

*Features Comparison

In order to rate each instrument a point system was developed to compare the various features of each instrument (Table SCI-4). The final rating will be instrumental in determining which component from the list will chosen for the mission. The feature comparison and rating can be found in Table SCI-5.

Point System Comparison Tool

Power: 1W = 1 pt. Mass: 1kg =1 pt. Size: $1m^3$ =1000 pts.

Data Rate: 1kbps = 1 pt. Swath width/FOV: @10km below 185km=1 pt.

Extras:
Pointing Required: 500pts. (scan platform & hardware) Antenna Size: $1m^2$ = 100 pts.

New Technology: 500 pts. (cost & reliability) Reliability: 1000 pts./year under 4 yrs.

Table SCI-4

Feature Comparison

Instrument	Power (W)	Size (m^3)	Mass (kg)	Data Rate (kbps)	FOV (km)	Extras	Rating
SAR	30	0.01	60	20	100	Antenna 10mx2m	2128.5
TM	300	1.54	239	850	185	--------	2929.0
RAM	25	0.03	30	30	185	Antenna .3m dish	143.3
RBV	130	0.081	59	50	185	1 yr. life	3320.0
CCDI	20	0.225	28	50	185	--------	323.0

Table SCI-5

The option selected is the Radar Altimeter Mapper (RAM). The fact that the instrument has the lowest rating is augmented by the fact that the Mars Geoscience Climatology Observer (MGCO) carries a similar instrument filling a similar science objective. A complete list of final picks for the mission instruments is found in Table SCI-6.

Instrument List

Instrument	Mass (kg)	Power (W)	Data Rates (kbps)	Size (m x m x m)
Gamma-ray Spectrometer (GRS)	6	10	1.3	.30 x .35 x .20
electronics package	9	-	-	.20 x .40 x.20
UV-Vis-IR Spectral Mapper (UVISM)	6	4	3	.40 x .40 x .31
Magnetometer	6	6	.2	.15 x .10 DIA
Radar Altimeter Mapper (RAM)	30	25	30	.25 x .40 x .30
UV Photometer (UVP)	5	6	.5	.15 x .15 x .10
Radio Science	5	18	1	.15 x .20 x .15
Hi Res IR/Adv Microwave Sounder (HRIR/AMS)	33	31	2000	.20 x .40 x .35

Table SCI-6

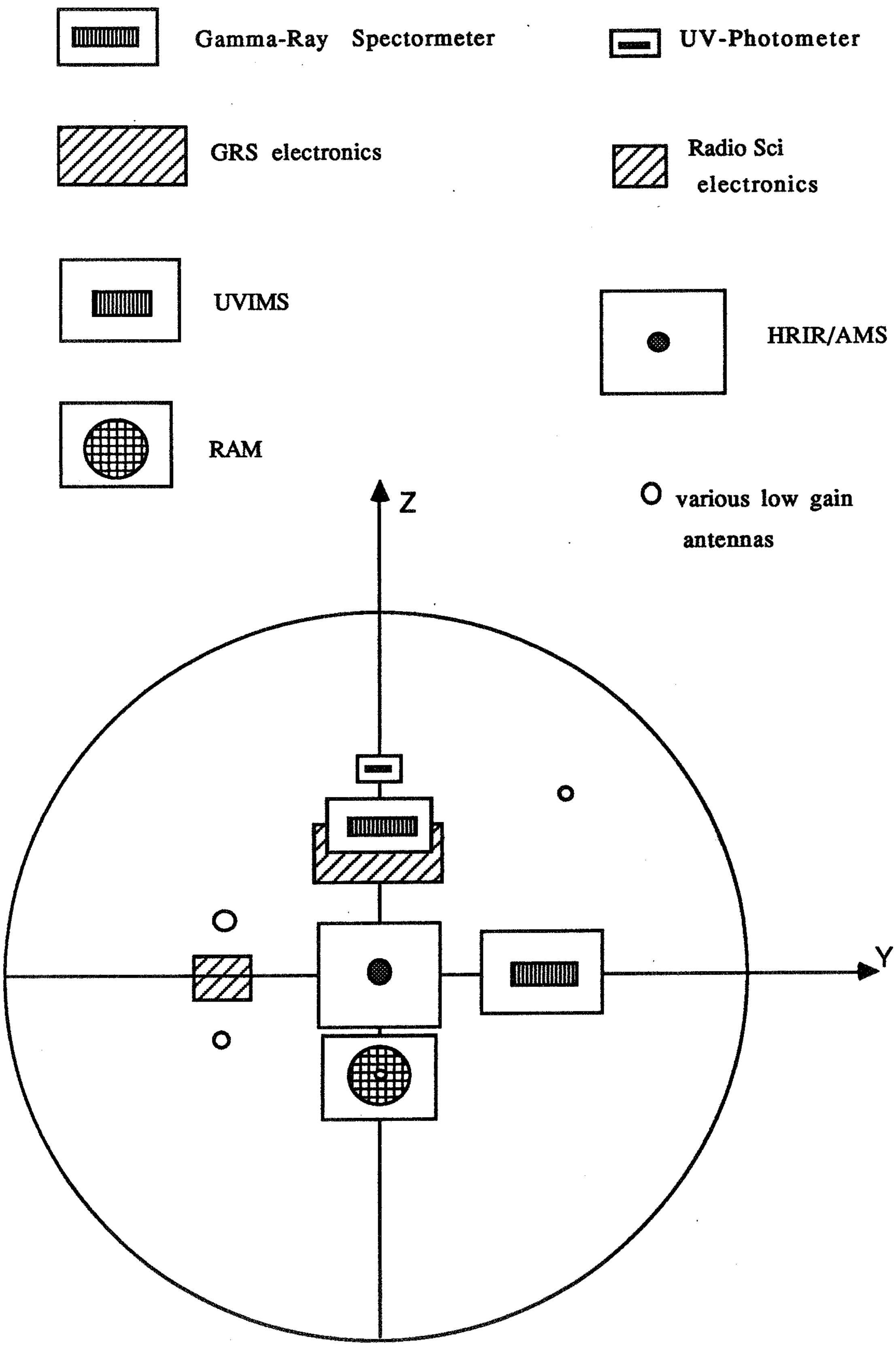
Instrument Placement

The first step in instrument placement involved the need for a scan platform. To perform the needed experiments to meet requirements the SCIS investigated the Integrated Platform Pointing and Attitude Control System (IPPACS). The IPPACS concept offers many advantages to the science group. This pointing system provides tight and accurate pointing as required by the instruments. IPPACS also effectively decouples the science instruments from the spacecraft by counter torquing gyros during any platform actuation. It adds flexibility to the EMTV satellite for other missions or science opportunities (MM II program p. 12). The SCIS proposed integrating the IPPACS package into the design of the EMTV.

Upon discussing the idea with Mission Management and Attitude and Articulation Control, the idea of IPPACS was removed from consideration. The rational being that scientific experiments on the EMTV were of low priority. The added weight, cost, reliability issues, and added complexity also helped to push the idea from consideration.

It was then decided to mount the instruments rigidly on the "planet-facing" surface of a stability platform on the satellite bus.

SCIENCE INSTRUMENTS AND PLACEMENTS



All instruments would be mounted with sensors and antennas pointing directly "downward" toward the planet surface. This placement gives the instruments an unobstructed view of the planet surface. Pointing other than at nadir would require actuation of the entire satellite bus or by pointing sensors and antennas. This configuration severely limits the usefulness of the SCIS.

Special Requirements

To reduce AACS burns to point instruments, some components were modified to accomplish multidirectional sensing. One case involves the HRIR/AMS, since both nadir and limb experiments need to be performed, a special movable aperture was fitted to the main instrument . This aperture allows for nadir and limb samples to be taken without actuating the satellite bus.

The magnetometer poses still another special problem. To receive meaningful data the magnetometer must be clear of the satellite's magnetic influence. To do this, two things should be done. First reduce the spacecraft's magnetic signature by building with non-magnetic materials. Second, isolate the instrument from the physical vicinity of the satellite (Corliss, p. 520). The first solution is referred to the structure subsystem-the analyst should keep this requirement in mind while designing the structure. The second solution is again requires the structural analyst to provide a suitable boom for this application. Requirements needed by the structure subsystem are as follows:

1. Use non-magnetic materials
2. Boom mount magnetometer (approx. 5m)
3. Point magnetometer sensor in planet direction

Shielding requirements were also submitted to the structures subsystem. For temperature control most satellite instruments have an operating temperature range between -20 and 80 degrees Celsius (Corliss, p. 354). But some advanced sensing instruments use cryogenic temperatures. Specific operating temperatures have not been determined at this point in the design process. Additional shielding from the Sun's electromagnetic radiation will also be required, but exact specifics have not been completely examined at this point.

Platform, Boom Placement

The SCIS worked closely with the Structures Subsystem to place the science instruments on the stability platform. An attempt was made to provide each instrument with an unobstructed Field of View (FOV) of the planet surface. This was accomplished by placing the instruments with all sensors pointing along the normal vector of the stability platform. The platform was then placed on the "downward" end of the satellite. This will point the instruments in a direction perpendicular to the planet surface at all times during the orbit.

A major design concern arose regarding the placement of the science platform. The platform is tucked away under the Aerobrake shield all during Earth-Mars transit and prior to to payload decoupling (refer to Structures subsection). This severely limits the use of the science instruments while in transit and prior to decoupling the payload decent module. Without some alternate solution, this prevents the remote sensing instruments from determining the status of the decent module landing site. A change in the mission plan may be a solution. The idea being to decouple the Payload Decent Module (PDM) upon reaching a low Mars orbit after the Aerobrake maneuver. While the PDM is still in the low orbit have the SCIS instruments power-up and monitor the landing site. When the site is given a 'safe' status the EMTV satellite bus will signal the orbiting PDM to begin decent sequence.

Another concern regarding the placement of the platform is that the instruments are placed in danger of being damaged when the PDM is pyrotechnically decoupled. A solution may involve placing a protective cover over the platform that can be jettisoned (non-pyrotechnically) after the PDM has been decoupled.

SCANNING AND POINTING REQUIREMENTS

Orbit

Closely paralleling the MGCO development, an orbit altitude of 350 km was chosen. This orbit will allow the SCIS instruments to provide excellent resolution and good coverage of the planet surface. To provide near constant global coverage a circular (e=0) orbit is essential. To meet the objective of global coverage, a near-polar sun-synchronous orbit would be the ideal choice.

Upon discussion with the AACS it was determined (using $\Delta v = \sqrt{2\mu/r\,(1-\cos\phi)}$) that changing the orbit plane to polar from an insertion orbit with inclination of 25.1° would cost too much in weight and fuel. A compromise resulted in the experimentation orbit would have the same inclination as the insertion orbit. The rational being that the weight penalty for the added fuel is too high. Plus the science coverage is not the primary role of the EMTV. A complete analysis of the orbit and ground coverage follow.

Using orbit radius = 3743 km (350 km altitude)
$\mu = 42821\ km^3/s^2$

Assumptions:	planet is spherical	orbit inclination = 25.1°
	orbit is circular	rotational period = 24:37:22.67
	mass of s/c negligible	swath width = 185 km

s/c orbit period: $T_{s/c} = 2\pi\sqrt{r^3/\mu} = 1.9314$ hr.

Angle of planet rotation/hour = 14.63°

Angle of planet rotation/ s/c orbit = 28.24°

Ground dist./ orbit period (along equator) = 1672.23 km

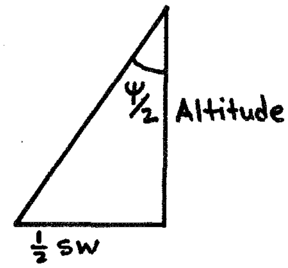
Shortest dist. between successive orbits = 1672.23 km sin 25.1° = 709.4 km

Orbits /Martian day = 12.75 this means there is a .25 orbit/day lag this translates into:

0.485 hr. 7.09° 420.1 km (along equator) 178.2 km between orbits

This means that in 3.98 Martian days the entire reachable surface will be mapped. In approximately 8 Martian days the surface can be redundantly mapped. Now with each ground trace 178.2 km apart we will chose a swath width of 1 nautical mile or 185 km.

Using a simplified triangle (neglecting the curvature of the planet) approach the sweep angle for scanners and the FOV angle for frame imaging sensors is given by:



$\tan \psi/2$ = Swath Width/(2 x Altitude)

so $\psi = \tan^{-1}$ 185/(2 x 350) = 29.6° or 30°

so the actual SW and FOV = 187.6 km

We will get approx. 5% overlap between adjoining ground traces. To get the same amount of overlap between successive 185 km x 185 km snapshots a simple rate was determined.

Ground path covered/orbit = $2\pi r$ = 21318.8 km

GPC/hr = 11038 km/hr need 185 km /snapshot

shots/hr = 11038/185 = 59.66 shots/hr. = 0.994 shots/min

or 1 snapshot every 1 minute necessary for 185 km x 185 km IFOV with approx. 5% overlap.

It was determined that with an inclination angle of 25.1° for the science experimentation orbit only 42.5% of the planet surface would be remotely sensed. This falls far short from the science objective total global coverage goal (Fig. SCI-4).

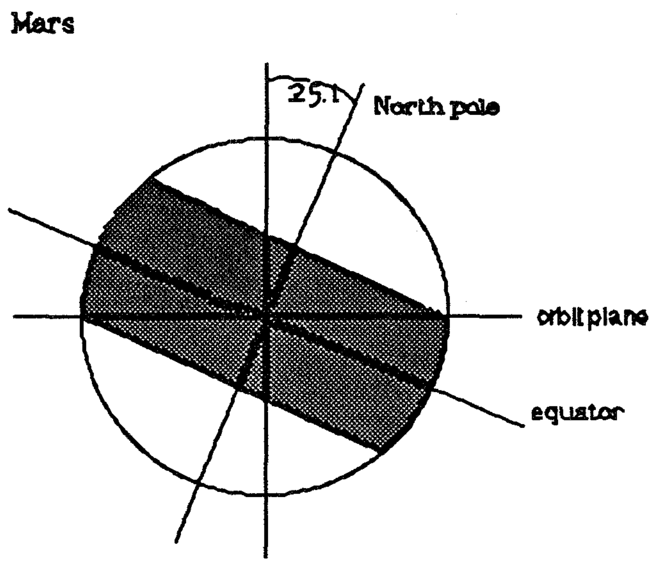
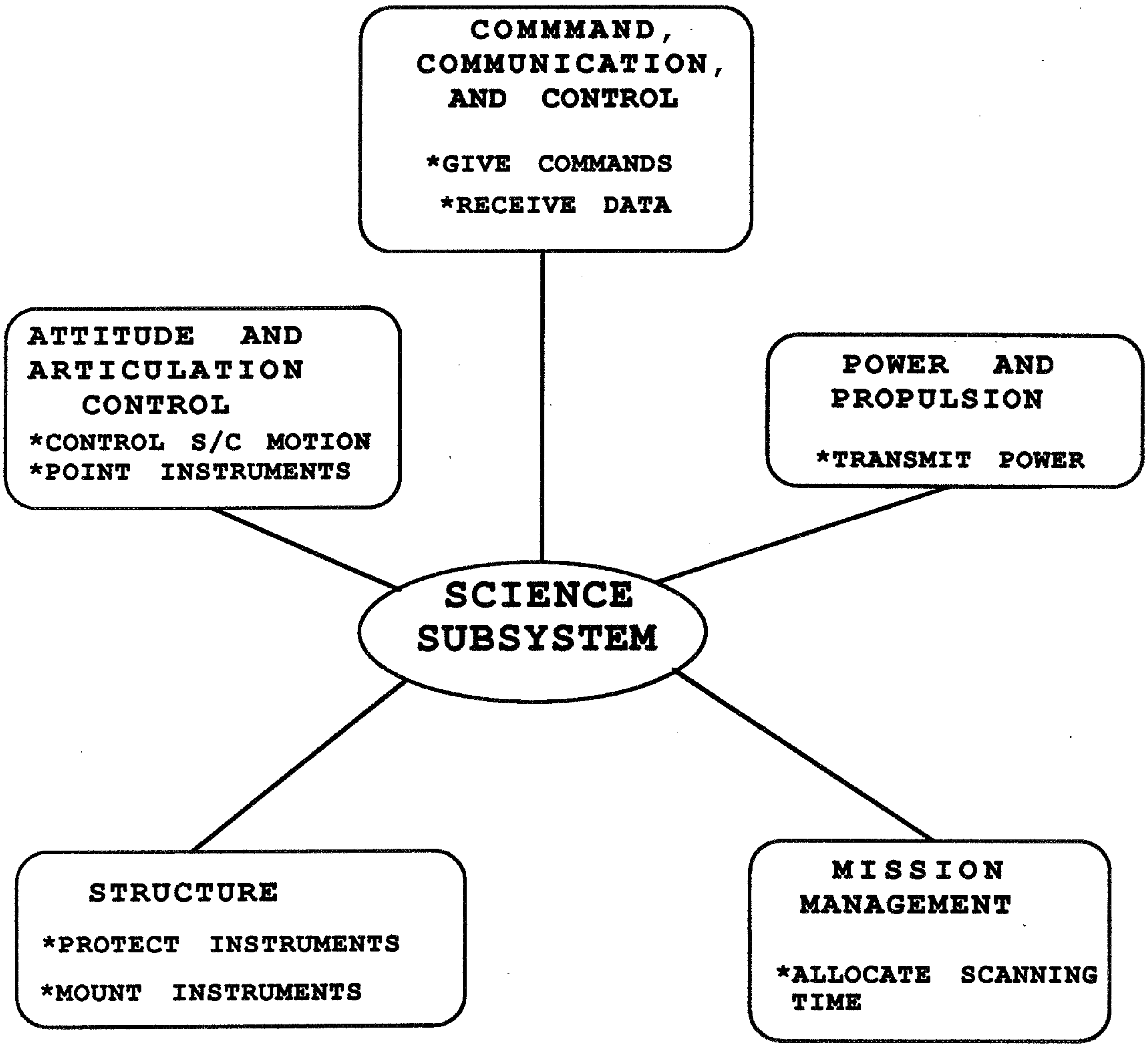


Fig. SCI-4

SYSTEM INTERACTION



EMTV satellite orientation

Since the science instruments need to view the planet surface to perform experiments - the satellite bus will be oriented with the instrument platform pointing directly "planet-ward" during the entire science experimentation mode.

The stability of the satellite bus must allow for accurate information gathering. From an altitude of 350 km the EMTV or the science platform must have a jitter (short term vibrations) rate of less than 7.5 μrad/sec to achieve clear readings and to minimize data correction(NASA sp-335 p.457).

The pointing of the EMTV and instruments should be as accurate as possible, errors on the order of 0.05 degrees are typical. Even so, the accuracies to which the pointing errors can be measured are far greater that the accuracies to which the pointing can be controlled. Therefore, it is possible to apply attitude and orbit corrections during processing to improve data quality.

SUBSYSTEM INTERACTIONS

Mission Management

Mission management needs to take orbital parameters and science requirements into consideration to determine the best method to incorporate science experimentation into the mission plan. Interaction is required to devise schemes to allocate time to complete various experiment sequences. The SCIS requests can be found in the scanning and pointing subsection of the science report. Final parameters should be found in Mission Management section of the Final Design Report.

Command, Communication and Control Subsystem (C^3)

-Prelaunch: The C3 subsystem will need to size tape recorders, digital/analog converters, data compression ratios, and multiplexers to handle all the raw data from the SCIS. The orbit and science experimentation sequence of the EMTV will determine the amount of data storage required. C3 will also have to program of the 'power-up' and experimentation sequences.

-postlaunch: The C3 will have to send command signals to begin the 'power-up' and experimentation sequences. It will also receive the raw data from the science instruments and format it before telemetering the information to Earth or Mars base.

Attitude and Articulation Control Subsystem (AACS)

-prelaunch: AACS should design system with SCIS requirements and needs in mind. Should attempt to incorporate needed spacecraft maneuvers for fulfilling SCIS requirements. Design system to best meet pointing and jitter requirements as specified by SCIS. (see AACS section to see methods used)

-postlaunch: Control EMTV attitude error and jitter to allow for acceptable data taking. Maneuver and hold EMTV in predetermined attitude in relation to planet surface. Adjust attitude of EMTV for additional experimentation if the need arises.

Power and Propulsion Subsystem (PPS)

-prelaunch: The PPS needs to take total power required by SCIS into account for the design of electrical power supply. PPS must also take into account eclipse time and the science experimentation sequence to size batteries. The total mass of the SCIS must be used to size boosters and fuel tanks.

-postlaunch: PPS must provide a continuous supply of electrical power to operate instruments. Must also provide for protection of instruments from power spikes and failures.

Structure Subsystem (Struc)

-prelaunch: place instruments on satellite in order to carry out science requirements. Mount instruments in a manner to protect them fro vibration and shock. Struc must incorporate shielding and thermal control as specified by the SCIS. Take the mass and instrument configuration into account when balancing the EMTV and determining moments of inertia.

-postlaunch: Keep instruments within the specified operational ranges using design methods.

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LARRY KIM
AEROBRAKE

INTRODUCTION

The mission objective is to deliver a Manned Mars Aircraft to the Martian surface. The delivery system is required to be easily operated and reliable with optimum performance, weight, and cost factors.

The mission is divided up into six subsystems which are Aerobrake, Structure, Power and Propulsion, Attitude and Articulation Control, Command and Data Control, Science and Radio Relay Instrumentation, and Mission Management. The purpose of this paper is to present the research and final design of the most optimum Aerobrake System.

This study of Aerobraking uses the atmospheric drag of a target planet to dissipate orbit energy and to finally circularize its orbit from a prior incoming trajectory. Aerobraking involves the trade-off study of avoiding the use of propellant masses which are normally carried along during the mission versus the mass of Thermal Protection System (TPS) and structural elements which are associated with the Aerobrake System (ABS). This is the trade-off study between propulsive stages and aerobrake stages which will be examined in further detail later in this study.

The ABS mission requirements are primarily to protect the instrument bus and payload from damaging heat during entry into the Martian atmosphere, to keep in constant communication with Command Data and Control during aerobraking, to communicate with the instrument bus for its safe terminal descent, and to provide a safe and reliable system to descend the payload to the surface of Mars. A final requirement is that the ABS be mass and aerodynamically efficient to maintain its total benefits over propulsive and aerocapture stages.

The ABS is a "complete spacecraft" in design which is integrated with the final mission spacecraft. For this reason integration with all other mission subsystems is mandatory and essential in acquiring the optimum aerobrake system. Before any initial design of the ABS, Mission Management and the ABS engineer must work together to provide atmospheric and trajectory parameters essential to aerobrake design. Clarification of the request for proposal is also reiterated during these conferences. Incorporated with the trajectory and orbit parameters requested is the integration with the Science and Relay Instrumentation Subsystem. Cooperation with the Science Subsystem, in part, determines the final orbit radius needed for correct instrument bus placement and operation. The Structures Subsystem is constantly in touch and working together with the Aerobrake Subsystem. The Structure Subsystem cooperation is essential in deriving working masses, structural dynamics, thermal protection materials, and with terminal descent structural rigidity and impact absorption. Ultimately, the complete layout of components is derived while working with the structures subsystem.

The Power and Propulsion Subsystem (PPS) is integrated into aerobrake design in initial aerobrake trajectory calculation. Using propulsion parameters of performance and trajectory axis, shield areas are optimumly computed. The Power Subsystem is also consulted during definition of power necessities in the ABS. The Attitude and Articualtion Control Subsystem integrates into the ABS by resolving attitude and articulation problems during aerobraking as well as during terminal payload descension. The last subsystem, Command and Data Control, integrates into the ABS by receiving data from the aerobrake during aerobraking as well as during payload descension. Computers selected by this subsystem command separation and retropropulsion during payload descension. This is a brief description of interactions of subsystems with aerobraking. Those interactions defined here are not the total range of interactions involved in the complete design of the ABS. During the design phase, to accomodate for any changes affecting the ABS, every subsystem is in touch with the Aerobrake Subystem.

TRADE STUDY OF CAPTURE SYSTEMS

There exists three different systems designed for orbit capture. They are the all propulsive system, the aerocapture system, and the aerobrake system. All three systems are designed to capture an orbit, hence it is necessary to select the proper system for the mission. This is a trade-off study between mass of propellants and mass of structures which must be examined closely. Due to the extent of space allotted for this report follows.

The final orbit capture system researched is the ABS. The ABS uses a low energy propulsive burn to reach the target planet, Mars, and uses a propulsive burn to be captured into an elliptical orbit about Mars. The spacecraft then performs multiple passes through the upper atmosphere utilizing atmospheric drag to decrease the elliptical orbit size until the desired apoapse altitude is achieved. The ABS then uses another small propulsive burn to circularize its orbit about Mars. The biggest disadvantage compared with the other two orbit capture systems is the longest time expended to achieve final orbit. This disadvantage is outweighed by the system's advantages. One advantage is the greatest mass delivery capability and efficiency with cost. The ABS is more reliable than the aerocapture system since it makes more passes through the atmosphere allowing it to make easy navigation adjustments. The aerodynamic heating is the lowest, therefore less mass is spent on TPS. Navigation requirements are low compared with the aerocapture system and moderate compared with the basic navigation system of the all propulsive system.

Seen clearly by the trade-off study shown here, the ABS for orbit capture is the most attractive and most efficient system for this mission.

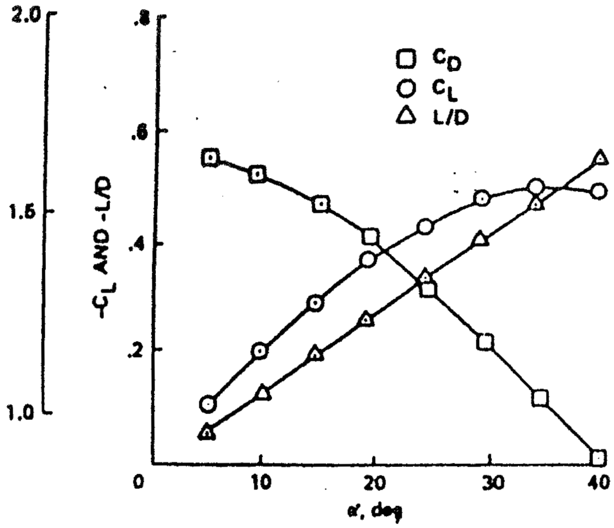
This paper now presents the complete layout of the ABS designed for the spacecraft delivery of the Manned Mars Aircraft . For illustrations of the component layout, look in the Structure Subsystem Report This reference is made to limit this paper to 100 pages. The complete component layout is followed by reports on each component and the method of attack in achieving the final design of the particular component.

THE AEROSHELL

The aeroshell is designed to deccelerate the payload at initial upper altitudes of reentry. Actual altitudes and speeds will be discussed later in the ABS mission timeline. The aeroshell's shape also provides aerodynamic lifting to allow for entry into the martian atmosphere after deorbit. The aeroshell structure is composed of aluminum ringed struts covered with an aluminum alloyed skin for maximum mass efficiency. Atop this skin is the thermal protection system which is comprised of a carbon-carbon material. This skin has an added safety feature of 2.0 to insure adequate temperature protection during reentry. A further study of TPS is included in the Structures Subsystem Report.The aeroshell is designed to withstand the upper altitude entry loads sufficiently where as not to damage the payload. This is done by distributing entry loads through the aluminum ring structure and into the payload compartment frame which evenly carries the load off of the aeroshell. The aeroshell interfaces to the circular payload structure at 25 points where the entry compression and shear loads are transferred from the aeroshell to the payload structure. Ten of these points are where explosive separation nuts are installed. These pyromechanical devices separate the aeroshell from the payload after descension through the upper atmosphere. Also at five points around the circular payload compartment are guide rails/rollers which are attached to the shell and payload. This gaurantess that the aeroshell is jettisoned in a smooth, controlled release .

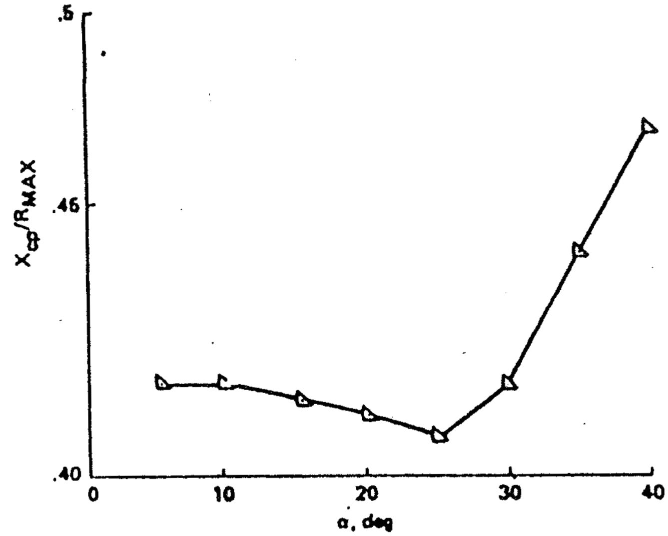
The aeroshell design is a 9.4 diameter, hemispherical,symmetric body shape. The symmetric body is designed with a fore half cone angle of 70 degrees for the best compromise between center of gravity and longitudinal stability. Since the aeroshell is symmetric the aft cone angle is -70 degrees. Using graphs 1a-1d, maximum angle of attack was calculated for this geometry. As seen in graph 1a L/D, drag coefficients,and lift coefficients were calculated for a range of angles of attack. Graph 1b shows the stability parameter versus angle of attack. The sharp decrease from 25 to 30 degrees indicates maximum angle of attack for stability lie in this range. Graphs 1c and 1d illustrate the quick deviation of mach number and of center of pressure after 25 degrees. This aeroshell has wake shadow of18m and has a much better reentry temperature distribution across its frontal surface due to its smooth blunt nosed design, as compared with an aerocapture aeroshell. One problem with an symmetrical aeroshell would be its lack of roll stability, but in this ABS mission the angle of attack will not surpass the 25 degree critical limit which would render the spacecraft unstable during the mission. Hence, at zero angle of attack the lift coefficient = 0.3 and the drag coefficient = 1.7. The L/D ratio at the critical angle of attack of 25 degrees = 0.35 which is optimum.The mass of

$C_D, C_L, L/D$ Vs α (Angle Of Attack)



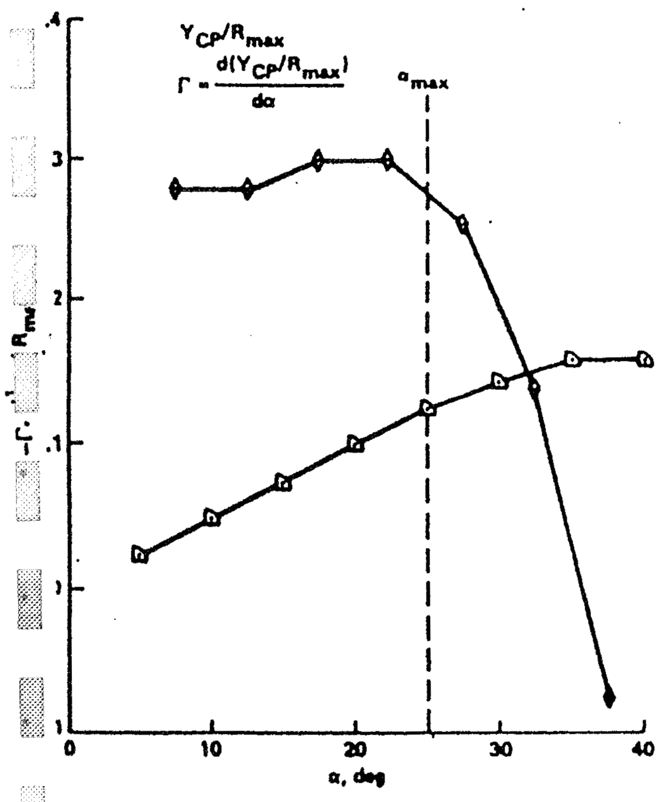
GRAPH 1a.

X_{cp} (Center Of Pressure) Vs. α



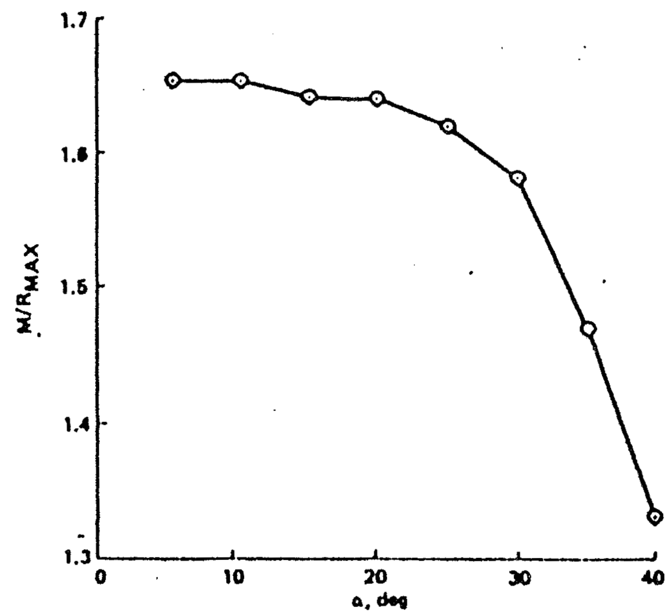
GRAPH 1c

Stability Parameter (Γ) Vs. α



GRAPH 1b

Mach # Vs α



GRAPH 1d.

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Aerodynamic characteristics of body in Fig. 2.

the aeroshell with the TPS and structure is 887.0kg. The total aerobraking stage mass is 5100.0kg. Therefore, the ratio of aeroshell to spacecraft mass is 0.10 or 10% which is very much more efficient than an all propulsive stage. This shield is optimumly designed in mass considerations. In conclusion, this aeroshell is stable up to angles of attack up to 25 degrees and has the necessary aerodynamic features to achieve the mission goal. The aeroshell has a protective shadow of 18m which is sufficient for this design. The aeroshell has adequate thermal properties which include safety factor of 2.0 in reentry temperature calculation, and is stable in the desired range of angles of attack. Stressing simplicity and reliability this aeroshell is relatively simple in construction and efficient in perfomance.

PROPULSION SUBSYSTEM.

The ABS consists of its own propulsion and control system. The propulsion and control system employed is used for the deorbit phase as well as for the terminal descent phase of the ABS mission. The components of the propulsion system are four Deorbit Controls System (DCS) engine modules, three Terminal Payload Descension (TPD) engines, and two system propellant tanks.

The four DCS engine modules perform the deorbit maneuver on command from the Central Aerobrake System Computer (CASC) which oversees the reentry and terminal payload descension phases of the mission. These four DCS engine modules also perform in flight attitude adjustments during reentry as well as during final touchdown. Placement of these four DCS engines are on the circumference of the payload structure each equally spaced and symmetrical on the structure.

Each of the four DCS engine modules is comprised of three thrusters which are a yaw/pitch pair and a roll thruster. The thrusters are chemical spontaneous catalytic monopropellant engines which are discussed in further detail in the Attitude and Articulation Control Subsystem report. it is sufficient to recognize that the four DCS engines perform attitude control upon command from CACS during the aerobraking,deorbit, and terminal descension mission phases. The redundancy of thrusters is a safety feature that will compensate for any critical valve loss during the mission.

The three TPD engines respond to CACS commands and will perform retropropulsion burns for terminal payload descension. CACS will throttle these engines according to sensor data it receives. Each of the three TPD engines deliver 2000 lb-thrust and the propellant is throttled by valves in the system. Specifications on these three TPD engines are in the Power and Propulsion Subsystem(PPS) report and should be reviewed for actual construction. One problem addressed here is the landing site deterioration from exhaust plumes. This is alleviated by installing 20 nozzles per engine to divert the exhaust plumes form the landing site.

The sizing of the two propellant tanks which fuel the DCS and TPD engines are in the PPS report. In this report their component placement is upon the payload structure inside the adapter ring. Placement of DCS engines,TPD engines, and tanks are shown in figure 6.

DYNAMICS AND CONTROL SUBSYSTEM

The CACS is the computing center for the ABS and there are many components which interact with this central brain. These components are Radar Altimeters(RA), Inertial Sensor System (ISS), Doppler Radar Altimeters (DRA), and an Onboard Decoder Amplifier Radio (ODAR). These active components control the ABS from aerobraking into final circular orbit to terminal payload descension.

The ISS is a system of four sets of principle-axis gyros and accelerometers , the fourth set being a redundant safety feature. These sensors provide data to CASC on velocity, altitude, and altitude rates during the deorbit and terminal descension phases.

The RA and the DRA are redundant devices for inflight calculation of altitude. These sensor data are superimposed with data from the ISS to gain more accurate inflight performance data. The location of the RA antenna is on the aeroshell providing an unobstructed view of the Martian surface. After the aeroshell is jettisoned, it is the DRA which provide the most accurate altitude data to CACS. The DRA's are placed under the payload structure without obstruction to the TPD engines.

All of the encoded data from the ISS, RA, and DRA's is sent to CACS. This transmission is done by the ODAR which is a low- gain radio. THe ODAR then decodes transmissions from CACS and implements commands for the two propulsion subsystems elements. The ODAR commands DCS engines on deorbit as well as TPD engines during terminal descension.

POWER SUBSYSTEM

The power system consists of the Power Distribution Assembly (PDA), batteries, and RAdio-Isotope Thermoelectric Generators (RTG). The power system is sealed in the payload adaptor which is atop the payload structure.

The PDA decodes commands from CACS and does the switching of electrical loads for the system. PDA also converts RTG energy to usable nominal voltage of 28. THe PDA switches RTG's from assorted system loads to the Batteries.

The four 24-cell nickel-cadmium batteries are each capable of storing in excess of 8 amps- hour. Fully charged, the batteries produce 1060 watts per hour. This along with the RTG constant power of 69 watts is sufficient for the aerobrake system power requirements of 60 to 70 watts per hour.

PARACHUTE SUBSYSTEM

In parachute selection, simplicity, and reliability are heavily stressed. The three parachutes are entirely fabricated of Dacron Type 52 and collectively weigh 290 lbs. Parachute dimensions are located in figure 6. Dacron Type 52 has a tested strength of 104 lb/in. The Kevlar parachute lines are tested to 900 lbs. Collectively , the parachute system has been tested in Earth atmosphere to mach 2.3 and dynamic pressure of

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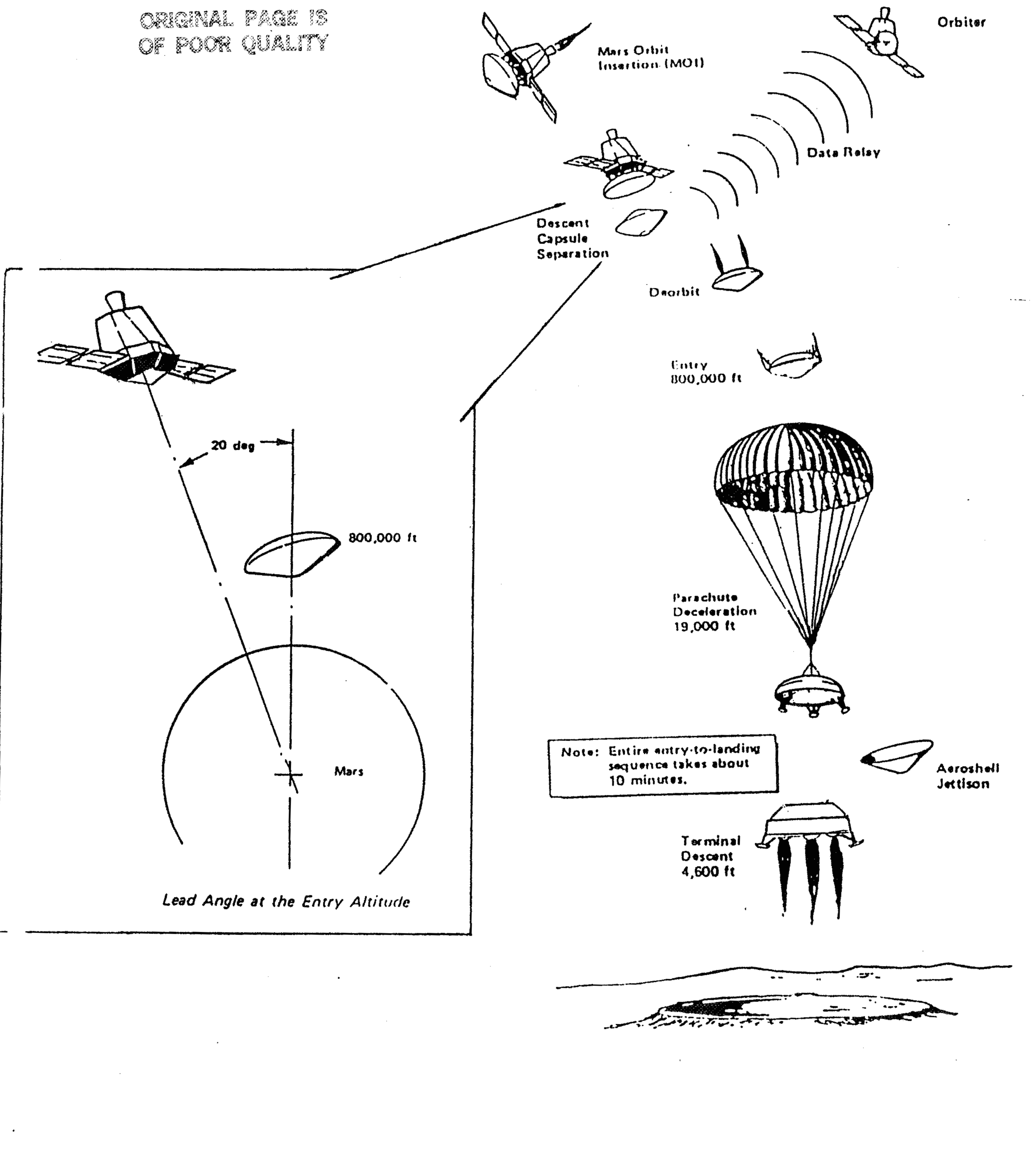

Lead Angle at the Entry Altitude

Mission Sequence

FIGURE 3

10.9 psf. This includes a safety factor of 1.25. The three parachutes are packed together into deployment bags and stored in a mortar can in the adaptor structure on the payload. The parachutes are ejected from the adaptor structure by pyromechanical charges upon CACS command.

On initial inflation attitude rates are extremely high, 96 degrees / sec, but are reduced to less than 25 degrees/seconds by the time aeroshell is jettison. The parachute aids in aeroshell jettison by keeping the payload structure aloft during separation.

LANDING GEAR

The landing gear,three absorbtion legs, are utilized to decrease touchdown loads. The three legs which are initially stored a the compressed configuration and have circular foot pads for terrain addhession in the event of landing on a incline. The touchdown loads are dissipated through the legs and into the payload compartment structure.

The dimensions on the legs are on figure. The legs are fabricated of a crushable aluminum honeycomb which absorbs load attenuation excellently.

FINAL AEROBRAKE SYSTEM MISSION TIMELINE (figure 3)

Phase I : Spacecraft arrives in Martian Elliptical Orbit and aerobraking begins. Initial mass of 10393.0 kg on start of aerobraking is decreased to 6409.70 kg after 20 aerobraking orbits. The initial elliptical orbit decays in eccentricity as the orbit intersects the upper Martian atmosphere. This initial elliptical orbit has semi-major axis of 44266km and periapse radius

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of 3600km. During aerobraking, CACS collects performance data and correspondingly commands burns to lower the orbit eccentricity. Upon obtaining circular orbit, 9.85 days after the start of aerobraking, the instrument bus and aerobrake system separate and continue orbiting Mars . After the instrument bus deciphers proper data for aerobrake system reentry maneuvers, the aerobrake system deorbits and begins reentry into the Martian atmosphere.

Phase II : After the CACS commands the aerobrake system separation from the instrument bus by pyrotechnical nuts, the CACS commands the ignition of the DCS engines for deorbit. Upon ignition of the DCS engines the ABS aligned the lead angle for entry into the upper Mars atmosphere. The lead angle is 20 degrees. At 3.2 hours after the DCS engines ignition, the ABS enters the upper Martian atmosphere at 800,000 ft. (244 km). CACS then activates and begins receiving data from the ISS and RA systems for guidance and control.

Phase III : When the ABS reaches 19,000 ft. (5.8 km), CACS commands parachute mortar ignition and .the parachutes are inflated. The parachute then aids in aeroshell jettison by keeping the payload structure aloft as the aeroshell falls away. Ten seconds after parachute inflation, the CACS commands aeroshell jettison. The pyromechanical shearing nuts are blown and the aeroshell falls away from the payload structure which is being held aloft by the parachutes. The payload then continues parachute deceleration.

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The CACS also at this time ignites the pin pullers on the landing gear, the three compression legs, and the legs extend and lock out to their terminal descent positions.

Phase IV : As the ABS descends by parachutes, the CACS gathers data for terminal descent and at 4600 ft. (1.4 km) the TPD engines are fired and the parachutes are detached. At this time the DRA is activated and sends altitude and velocity data to CACS. Then from this altitude the CACS commands throttling of the payload module according to data sent by RA, ISS, and DRA systems until final touchdown. Final touchdown velocity is constant from 55ft. at 8 fps and by final touchdown the TPD engines will have fired for 30 seconds.

Phase V : Upon touchdown, the CACS receives status signals from the ODAR on the payload module and pinpoints its location. This information is then relayed to the on Mars space station. The entire entry to landing sequence consumes 10 minutes.

CONCLUSION

The final design of ABS stresses reliability, efficiency, and costs in all aspects of design. Where critical design decisions had to be made, safety factors were employed. Perhaps the most difficult part of this design is the testing of the critical subsystems of the ABS. Collectively, the ABS is an entirely separate and complete spacecraft that requires all the testing procedures required of a spacecraft design. Once again, when difficulty arose in testing procedures, the best method of assurance was utilized and that is the input of safety factors.

Although this system was designed for a payload of a Manned Mars Aircraft, nothing in its design precludes it from any other mission. The design is reliable, efficient, and has nothing in its design to prevent it from being used for many years to come.

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Project A.C.R.O.N.Y.M.

Awesome Craft - Really Overshadowing Next Year's Mission

E-MTV

EARTH-MARS TRANSPORT VEHICLE

Command-Control-Communication Subsystem

C^3 SPECIALIST: Sisir R. Kudva

C^3 REQUIREMENTS

The C3 Subsystem basically consists of two major components: the computer and the communications equipment. The general system requirements are given at the beginning of this report in the Mission Management section. The specific requirements for this subsystem are as follows:

- Collect telemetry from other subsystems.

  In this case, the main subsystems interacted with are the Power and Propulsion System, the Attitude andArticulation Control subsystem, and the Science subsystem.
- Send telemetry to ground.

  All data collected from pertaining subsytems will besent to ground. In the first part of the mission, the telemetry will all be sent to Earth. In the latter part of this mission, after aerobrake maneuvers, the telemetry could also be sent to the Mars base. After the aircraft has been assembled, all data necessary to the operations of the aircraft will be sent directly to the aircraft.
- Send commands to subsystems.

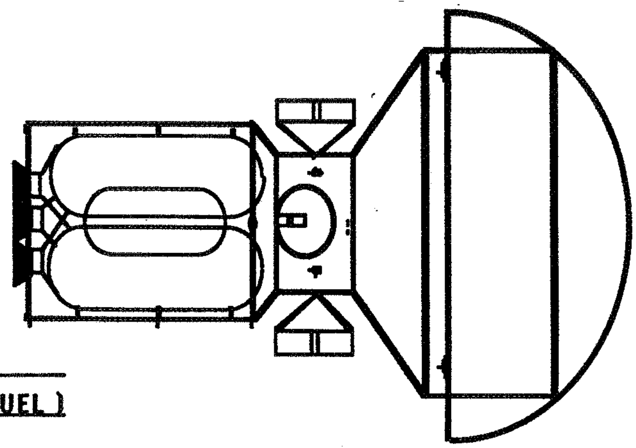
  The commands will originate from one of two places. First, the systems software will be sending commands to various subsystems at the proper time as seen by the program. Second, ground control (Earth or Mars) may, if necessary, override the system software and send new commands for various subsystems. These commands will first be processed by the computer and then sent to the proper components.
- Power switching for subsystems

  Each subsystem will have different power requirements at different times. The subsytems requiring power are the C^3 subsystem, the AACS subsystem, the STR subsystem, and the SCI subsystem. The SCI subsystem will be inactive in the former stages of the mission.
- Power from PPS

  The C^3 subsystem will be receiving its electical power from the PPS subsytem.
- Control outputs
  - Power switching
  - Telemetry to ground
  - Commands
- Inputs
  - Telemetry
  - Power requests
  - Commands from ground

JEFF BRADSHAW
GROUP 5

MASS OF SPACECRAFT AND INERTIA TENSORS AT DIFFERENT MISSION STAGES



AT LAUNCH

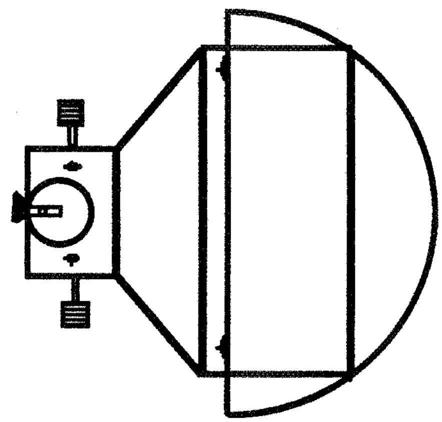
MASS = 22152 KG.

$$I = \begin{bmatrix} 67503 & 341 & -1 \\ 341 & 26697 & -21 \\ -1 & -21 & 357063 \end{bmatrix}$$

DURING TRIP TO MARS (LESS FUEL)

MASS = 10920 KG.

$$I = \begin{bmatrix} 44622 & 341 & -1 \\ 341 & 187843 & -21 \\ -1 & -21 & 192899 \end{bmatrix}$$



DURING AEROBRAKE STAGE

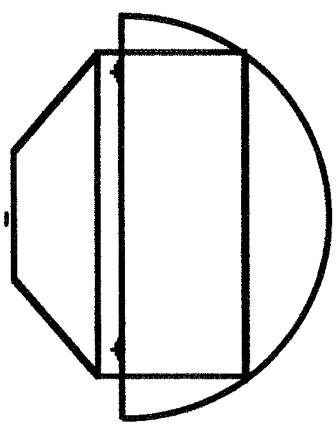
MASS = 7666 KG.

$$I = \begin{bmatrix} 36540 & 341 & -1 \\ 341 & 68519 & -21 \\ -1 & -21 & 71234 \end{bmatrix}$$

DESCENT CAPSULE

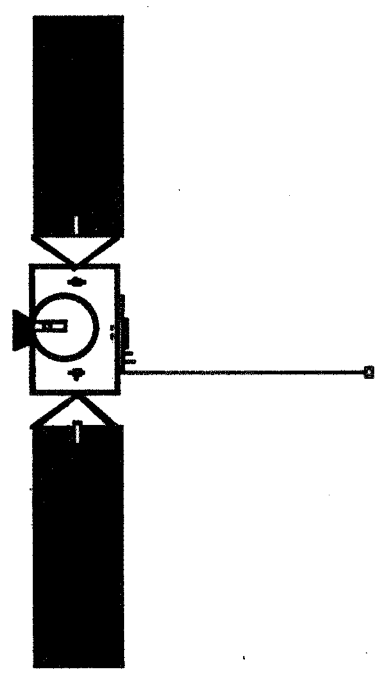
MASS = 4600 KG

$$I = \begin{bmatrix} 33068 & 337 & 0 \\ 337 & 26697 & -24 \\ 0 & -24 & 29403 \end{bmatrix}$$



MASS OF FINAL ORBITTING BUS = 1146

$$I = \begin{bmatrix} 2579 & 22 & 102 \\ 22 & 2344 & -19 \\ 102 & -19 & 1548 \end{bmatrix}$$



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Mars
Aircraft
Receptacle with
Technical
Instruments,
Aerobraking and
Navigation

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Table of Contents

Introduction

In response to the Request for Proposal and Preliminary Design of a Manned Mars Aircraft Delivery System, we, the members of the M.A.R.T.A.I.N. design team, have prepared this report to present our design of a spacecraft that is capable of delivering from low Earth orbit to the surface of Mars a man-rated aircraft. We belive that our design is an innovative yet flexible and reliable response to the problem stated in the R.F.P.

Certain requirments were seen to be global (i.e. pertaining to all subsystems) while others were seen as only applying to a specific subsystem. These subsystem specific requirements will be delt with in the individual subsystem chapters. Global requirements for the M.A.R.T.A.I.N. design are:

- Spacecraft will consists of a reentry system and a Communications/Science Satellite
- Use of Aerobrake maneuver
- Minimize use of Shuttle and on orbit assembly
- Retrievable by a remote manipulation device
- Able to perform different missions
- Lifetime of greater than four years
- Use of off-the-shelf hardware

- 1998 materials and technology cut-off
- Use of Artificial Intelligence
- Fulfillment of mission science objectives
- Optimize reliabilty, simplicity, low mass, and low cost
- Four spacecraft will be built, three will fly
- Subsystems
    - Aerobrake
    - Attitude and Articulation Control
    - Command and Data Control
    - Mission Management, Planning and Costing
    - Power and Propulsion
    - Science
    - Structure

<u>Mission Management, Planning and Costing</u>

I. Requirements:

The following is a list of requirements that apply to the MMPC subsystem. Each requirement is followed by the page number or numbers on which it is addressed.

From the RFP:

1.) Deliver a manned aircraft to Mars between 2005 and 2010. p.1-2,1-3, 1-6,1-7,1-8.

2.) Two components: payload reentry system and instrument bus. p.1-3.

3.) Subsystems: p.1-2,1-3,1-5,1-9.
   a.) Aerobrake (including orbit capture, reentry, and detachment)
   b.) Structure (including materials, design, thermal control)
   c.) Power and Propulsion
   d.) Attitude and Articulation Control
   e.) Command and Data Control
   f.) Science and Radio Relay Instrumentation
   g.) Mission Management, Planning and Costing

4.) Delivered to orbit by Space Shuttle and assembled in orbit. p.1-2,1-10.

5.) Be able to perform several possible missions. p.1-10.

6.) It should not use techniques available after 1998. p.1-10.

7.) Lifetime of at least four years. p.1-10.

8.) Stress simplicity, reliability, minimum mass and low cost. p.1-6,1-7.

9.) For cost estimating assume four spacecraft will be built. p.1-5.

Derived requirements:

1.) Calculate a trajectory to deliver the spacecraft to Mars. p.1-6,1-7,1-8.

   a.) Leave from low Earth orbit. p.1-2.

2.) Calculate an orbit around Mars for the instrument bus. p.1-5,1-6.

a.) Two satellites will be used to provide continuous coverage of the aircraft on Mars. p.1-5,1-6.

3.) Make an estimate of the cost of such a mission. p.1-5.

Requirements from other subsystems

1.) Satellite must pass below an altitude of 800 km. p.1-5,1-9.

II. Technical Approaches:

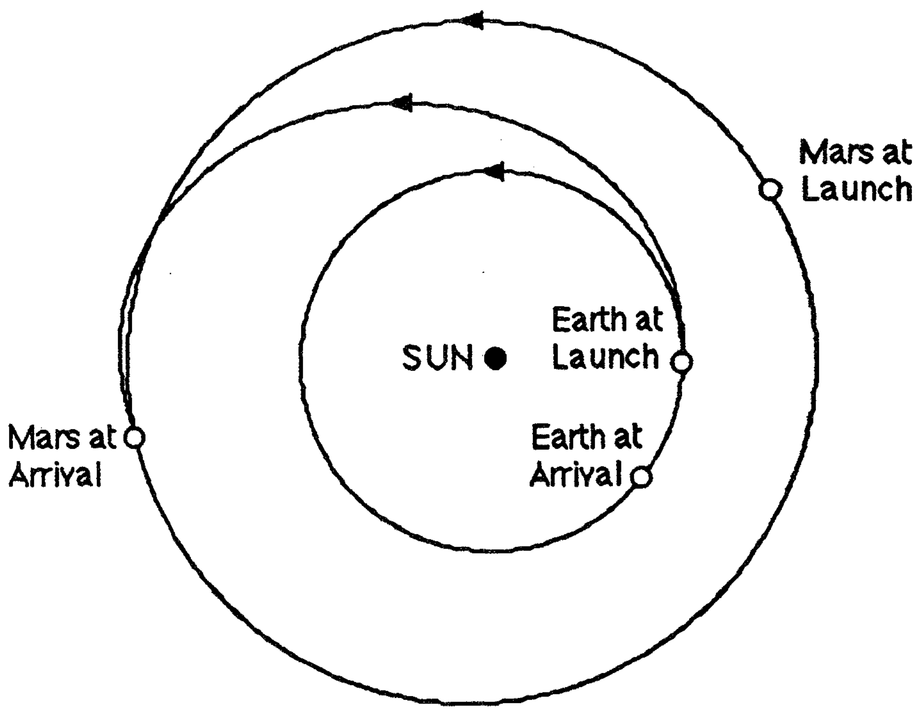

Fig. 1.1 Earth-Mars Trajectory

The trajectory chosen for the flight to Mars is a direct flight using a chemical high thrust system. The geometry of the mission is shown in Fig. 1.1. The spacecraft will be assembled in low Earth orbit. A single propulsive burn will be used to exit a circular orbit of 250 n.m. (463 km). This burn will take place on Oct. 14, 2009, and will require a ΔV of 3.628 km/sec. After a flight of 325.6 days, the vehicle will rendezvous with the planet Mars on Sept. 4, 2010. An aerobrake maneuver requiring 23.21 days will place the spacecraft into orbit around Mars. Starting the aerobrake sequence requires a ΔV of 615.1 m/sec. After the aerobraking is completed, a ΔV of 161.7 m/sec circularizes the orbit. A complete description of the aerobraking maneuver is given in the AERO subsystem report. The total ΔV to place the payload reentry system in Martian orbit is 4.405 km/sec.

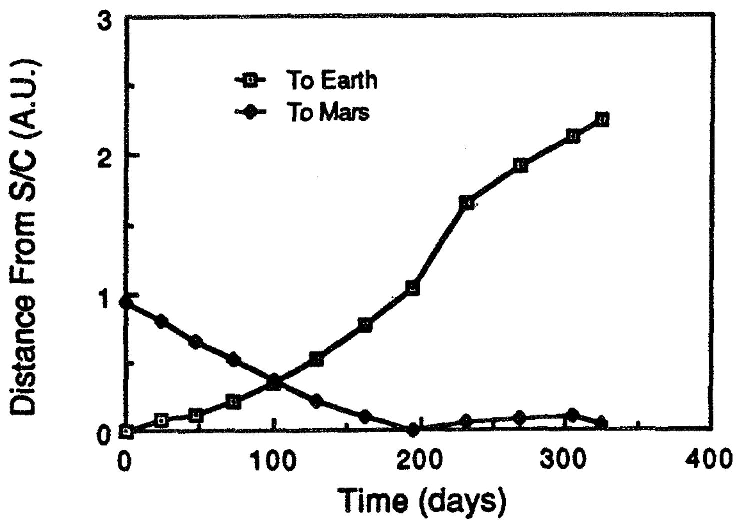

Fig. 1.2 Communications Distance

The distance from the spacecraft to Earth and Mars during the flight was calculated for the CDC subsystem. To reduce antenna size, the spacecraft will talk directly to Earth only during the beginning of the mission. When the distance from spacecraft to Earth is equal to the distance from the spacecraft to Mars, the antenna pointing will be changed. During the latter part of the mission, the spacecraft will communicate with the base on Mars. Fig. 1.2 shows the results of the distance calculation. The maximum communications distance was found to be 0.4 A.U. The pointing change will take place on Dec. 30, 2009.

During the aerobrake maneuver, a satellite will be released from the spacecraft into Martian orbit. This will occur after 23.16 days of aerobraking.

At separation, the satellite will be in an orbit with a semimajor axis of 6437.5 km and periapse radius of 3446.1 km. At apoapse of the orbit, a burn will be made

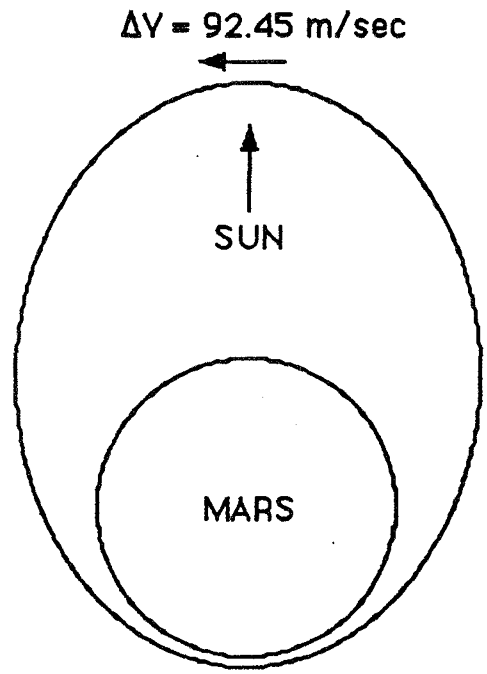

Fig. 1.3 Satellite Orbit

to place the satellite in its final orbit. The required change in velocity is 92.45 m/sec. The satellite's final orbit has the following dimensions:

semimajor axis = 6347 km
periapse radius = 3694 km
periapse velocity = 4.07 km/sec
apoapse radius = 9000 km
apoapse velocity = 1.67 km/sec
eccentricity = 0.418
period = 4 hours 15 minutes 12.55 seconds
inclination = 45.70495 deg

An inclination of 45.7 deg was chosen to make the orbit Sun-synchronous. A Sun-synchronous orbit precesses at a rate which causes the semimajor axis to always point toward the Sun. Apoapse of the orbit will be on the sunlit side of the planet. The satellite will therefore spend the majority of the time above the day side of the planet.

The cost of the mission was computed based on the masses of the various subsystems. A computer program which takes masses and estimates the cost to design and build a spacecraft was employed. The program yielded the following results. The non-recurring cost to design and develop the spacecraft is $1.3814 billion. The unit price to build each of the spacecraft is $446.7 million. This gives a total price tag of $3.1682 billion to design and build four spacecraft .

The following is a list some of the information required by the other subsystems. AERO: Arrival velocity at Mars. PPS: ΔV, satellite time in shadow. AACS: Spacecraft orientation. CDC: Communication distance, antenna pointing. SCI: Satellite altitude.

III. Technical Problem Areas:

Several problems had to be addressed in the determination of the satellite orbit. Phobos, the inner of Mars' two moons, has an orbit radius of 9300 km. Apoapse radius of the satellite orbit was chosen to be 9000 km to avoid the Martian moons. The Science subsystem dictated that the periapse of the orbit must be below an 800 km altitude. The altitude at periapse was chosen to be 300 km to maximize the time below 800 km while staying well above the Martian atmosphere. Due to communication considerations, the orbit was made Sun-synchronous with apoapse on the sunlit side.

Continuous contact must be maintained with the aircraft in operation on Mars. It is assumed that two aircraft and two satellites have been delivered.

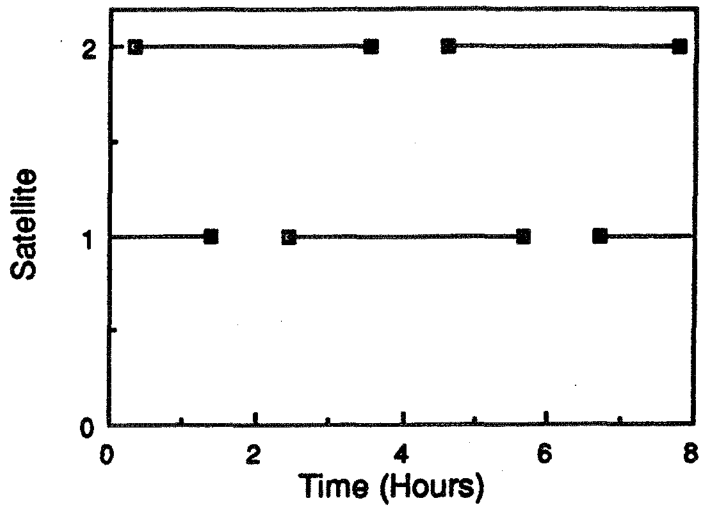

Fig. 1.4 Satellite Coverage

The two satellites will be placed in opposite ends of the same orbit. When one satellite is at apoapse of the orbit, the other will be at periapse. Together the satellites will maintain continuous contact with the aircraft. Fig. 1.4 illustrates which of the satellites is overhead at a given time. The bars represent when a satellite is in the part of its orbit above the sunlit side of Mars. The aircraft, which will be operating only during daylight, will have at least one satellite overhead at all times.

IV. Trade Studies:

Three types of interplanetary trajectories were investigated to deliver the spacecraft to Mars. The first type was a direct flight using a chemical high thrust propulsion system. The second type also used chemical propulsion but included a flyby of Venus in an effort to reduce the amount of ΔV required. The final type of trajectory used a low thrust propulsion system.

The criteria used to choose a type of trajectory were simplicity, reliability, minimum mass and low cost. Minimum mass requires that the ΔV be small. Low cost means that the length of the flight should be as short as possible. In general, the direct flight option is the simplest and has the shortest flight time, but has the highest mass. The Venus flyby option should yield a lower ΔV than the direct flight. A Venus flyby has several disadvantages. The flight time is usually longer than for a direct flight. Heat shielding would increase the mass of the spacecraft. A low thrust propulsion system has the lowest propulsive mass and would further reduce mass by reducing reinforcement needed for the spacecraft. Flight time for the low thrust option is much longer than those of the first two options, and it does not have the proven record of the high thrust systems. The low thrust system would not use an aerobrake to enter Mars orbit, thus violating a one of the requirements for the mission.

Option	Direct Flight	Venus Flyby	Low Thrust
ΔV (km/sec)	3.628	3.805	N.A.
Length of Flight (days)	325.6	302.0 ?	~730
Aerobrake	YES	YES	NO
Tested	YES	YES	NO

Option Selected

Fig. 1.5 Trajectory Trade Study

The trade study of types of trajectories compared numbers from computed trajectories. First of all, the low thrust system was ruled out because it did not employ an aerobrake system to enter Martian orbit. Trajectories for the direct flight and the Venus flyby were computed. The results are shown in Fig. 1.5.

The flyby trajectory yielded a shorter flight time than this particular direct flight. On the other hand, the ΔV for the direct flight turned out to be lower. It was felt that reducing ΔV was more important than reducing mission length. The direct flight option is therefore the optimal solution in this case.

A computer program called MULIMP was used to compute optimum direct flight trajectories. The analysis covered the period of 2005 to 2010. The

Date	Sept. 1, 2005	Sept. 22, 2007	Oct. 14, 2009
ΔV (km/sec)	3.855	3.737	3.628
Length of Flight (days)	401.3	368.6	325.6
Earth-Mars Distance at Arrival	2.609	2.469	2.234
Earth-Sun Angle at Arrival	3.03	13.4	24.3

Option Selected

Fig. 1.6 Direct Flight Trade Study

program was used to calculate trajectories with the lowest required ΔV. The computed ΔV did not include that needed to enter Martian orbit. During this period of 2005-10, three optimal trajectories were found. These three trajectories were compared with respect to ΔV, length of flight, Earth-Mars distance at arrival, and Earth-Sun angle at arrival. Fig. 1.6 shows the values of these parameters for the three trajectories. The chosen date, Oct. 14, 2009, is the best in all four categories.

Four different types of orbits were investigated for the satellite. The four types are polar and non-polar low circular orbits, synchronous orbit, and an

Option	Low Orbit Polar	Low Orbit Non-Polar	Synchronous Orbit	Elliptical Orbit
Continuous Contact	NO	NO	YES	YES
Science Instrument Range	YES	YES	NO	YES
Global Coverage	YES	NO	NO	YES

Option Selected

Fig. 1.7 Satellite Orbit Trade Study

elliptical orbit. The science requirements of the satellite are that it must pass below an altitude of 800 km for certain instruments to function. In addition, the orbit should cover as much of the planet as possible. For safety reasons the satellite must maintain constant contact with the aircraft. Fig. 1.7 lists whether each option meets each of the requirements. The only orbit that meets all the requirements is the elliptical orbit.

V. Implementation Plan:

The following is a timeline for a possible mission using the trajectory described previously.

Oct. 1, 2009	Launch of first Heavy Lift Vehicle
Oct. 2, 2009	Launch of second Heavy Lift Vehicle
Oct. 2-14	On-orbit assembly of spacecraft at Space Station
Oct. 14, 2009	Departure from Earth orbit
Dec. 30, 2009	Change in antenna pointing
Sept. 4, 2010	Beginning of aerobrake maneuver
Sept. 27, 2010	Release of satellite into Martian orbit, completion of aerobraking, and descent to surface after survey of landing site

In summary, the design given meets all but one of the requirements. It was found that the spacecraft could not be carried into orbit by the Space Shuttle. The justification of using the Heavy Lift Vehicle is given in the Structure subsystem report. The spacecraft will be able to perform different missions as long as the needed ΔV does not exceed the propellant carried. All of the techniques described are available now or will be available before 1998. Nothing in MMPC design will limit the spacecraft lifetime to less than four years.

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Science and Radio Relay Instrumentation

Monica M. Doyle

In addition to delivering the Mars aircraft to the surface of the planet, the MARTIAN mission will deploy two identical satellites, injecting them into two distinct highly eccentric orbits as described in the section entitled "Mission Management, Planning and Costing". The selection and coordination of these instruments is the task of the Science and Radio Relay Instrumentation subsystem. The science instruments were chosen only after careful consideration of the mission science objectives, research of past missions to identify candidate instruments and comparative studies of these candidates.

From the Request for Proposal, we have identified these global requirements as pertaining to this subsystem:

- the lifetime of the satellite, like that of the spacecraft itself, should be at least four years

- the design of the satellite will stress simplicity and reliability at a minimum cost while fulfilling the mission science requirements

- the technology proposed must be available by the year 1998.

From the document entitled "Mission Science Objectives", we obtained the general and specific science subsystem requirements. The general objectives of the MARTIAN science program will be:

- gain an insight into the origin, evolution and present state of the universe

- obtain a better understanding of our own planet through comparative studies

- determine the relationship between the evolution of the solar system and the appearance of life.

To meet these general goals, six specific subsystem requirements are outlined.

These are:

- determine the elemental and mineralogical composition of the Martian surface

- determine the distribution, abundance, sources and concentrations of volatile materials and dust

- define the global gravitational field

- measure the global surface topography

- examine the structure and circulation of the Martian atmosphere in detail

- determine the nature of the planet's magnetic field.

Tables 2.1 through 2.6 outline the advantages and concerns associated with each of the candidate instruments corresponding to the six specific science requirements for the MARTIAN mission. Below each table is a brief description of the instrument(s) chosen to fulfill that particular requirement and be carried on each of the MARTIAN orbiters.

Table 2.1

Req.:determine the elemental and mineralogical composition of the surface on a global basis

Candidate Instrument	Advantages	Concerns
High Resolution Imaging Spectrometer	highly programmable, high resolution	massive, high power requirements
Gamma Ray Spectrometer*	proven successful on Apollo missions	possible interference from S/C magnetic field
Near Infrared Mapping Spectrometer (NIMS)*	detailed spectroscopic characterization, combines spectroscopic and imaging capabilities in one instrument	low spatial resolution

*instrument chosen

The gamma ray spectrometry experiment on the MARTIAN orbiter is designed to obtain information on the abundance of elements present on the surface of the planet. These elements will be identified by the characteristic wavelength of the gamma rays they emit when bombarded by neutral particles. Elements that can be detected by this instrument include potassium, titanium, thorium, iron, uranium, silicon, oxygen, hydrogen and carbon. The instrument may be shielded from interference by the satellite magnetic field through the use of a double coaxial cable between the photo tube through which the energetic particles enter and the charge amplifier where the sensed energy is amplified and transformed into a digital signal to be sent to the data control system.

NIMS combines the capabilities of spectroscopy and imaging in a single instrument. Spatial coverage of the surface is derived from a scanning telescope and spectral coverage from the spectrometer. Composition of both atmosphere and surface will be examined by observing the infrared radiation from each.

Table 2.2

Req.:determine the distribution, abundance, sources and concentrations of volatile materials and dust

Candidate Instrument	Advantages	Concerns
Ultraviolet Photometer*	proven successful on Pioneers 10 & 11	poor resolution at large distances
Dust Detector*	provides broad spectrum of information incuding size, speed, charge and mass of dust particle	only examines particles at S/C altitude; provides no info about particles closer to surface
Global Ozone Monitering Radiometer (GOMR)	ultraviolet and infrared dual capabilities	massive, very high power requirements

The ultraviolet photometer will be used to measure the scattering of ultraviolet light or emission of ultraviolet light from interplanetary hydrogen, helium and dust from the atmosphere of Mars. The data gathered from these experiments will be useful in determining the abundance and distrubution of these volatile materials. Knowledge pertaining to the amount of helium in the interplanetary region is also expected to shed more light on the origin and evolution of the solar system.

The dust detection experiment is designed to gain an understanding of the characteristics of dust particles in and around the Martian atmosphere. The instrument is comprised of a set of grids, electron and ion collectors and circuitry to determine the mass, impact speed and charge of a particle entering the unit.

Table 2.3

Req.:define the global gravitational field

Candidate Instruments	**Advantages**	**Concerns**
Radio Science (in communication with Mars base)*	relatively simple	must account for Doppler shift when passing over surface transmitter
Radio Science (in communication with Earth)	proven successful on many NASA missions	requires radio communication with Earth (in addition to that with Mars base)

The radio science instrument is part of the celestial mechanics experiment to be performed by the MARTIAN orbiter. The object of this experiment is to define the gravitational field on a global basis. In this case, the measurements will be made by the satellite itself as it orbits the planet in communication with the Mars base. The radio system serves as a sensor for small gravitational perturbations on the orbiter trajectory which, in turn, are used to infer the structure of the gravitational field of Mars. This experiment will offer information concerning the density and mass distribution within the planet.

Table 2.4

Req.:measure the global surface topography		
Candidate Instrument	**Advantages**	**Concerns**
Radar Mapper*	proven successful on Pioneer Venus mission	moderate resolution
Solid State Imaging	high spatial resolution	limited spectral information, massive, high data rate

The radar mapping instrument will be used to obtain a topological map of the Martian surface. Surface heights along the suborbital trajectory are derived from the observation of a radar echo and are generally accurate to approximately .15 km. From this data, a good estimate of global topography can be obtained. Furthermore, information on electrical conductivity and meter scale roughness on the surface can be derived from these observations.

Table 2.5

Req.:examine the structure and circulation of the atmosphere in detail		
Candidate Instrument	**Advantages**	**Concerns**
Atmospheric Sounder*	high vertical resolution	relatively high power requirements
Ultraviolet Spectrometer*	high spatial resolution	retains capacities up to considerably large distances then begins to lose resolution
Photopolarimeter	very high spatial resolution	loses capacities at critical distances less than those of the UV Spect.

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The atmospheric sounding instrument on the MARTIAN science orbiter will provide a direct measurement of the temperature profile in the Martian atmosphere with extremenly high vertical resolution.

The ultraviolet spectrometer experiment is designed to determine the characteristic properties of the Martian atmosphere via a spectroscopic analysis of ultraviolet light scattered or emitted by clouds and atmospheric gases. Each gas in the atmosphere has its characteristic emissions and the flourescence in the atmosphere caused by the absorption of ultraviolet radiation has its peculiar characteristics dependent upon the elements present.

Table 2.6

Req.:determine the nature of the magnetic field.

Candidate Instrument	Advantages	Concerns
Flux Gate Magnetomater*	proven successful on both planetary and Earth satellites	possible interference with satellite magnetic field
Magnetospheric Currents and Fields Detector (MAG)	broad range of magnetic field sensativity	massive, high data rate, possible interference from satellite magnetic field

The magnetometer experiment on the MARTIAN orbiter will be used to determine the nature of the Martian magnetic field. The instrument is designed to produce a voltage that is linearly proportional to the sensed magnetic field. This voltage is then converted into a digital signal to be sent to the data control system.

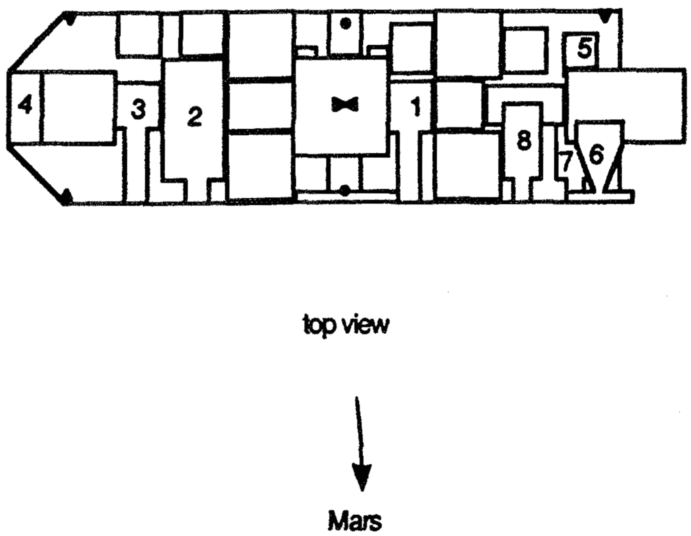
One method of stabilization for the orbiter will be the use of two gravity gradient booms of approximately 10 meters in length and oriented in opposite directions along the gravity vector. This will be discussed in detail in the section entitled "Attitude and Articulation Control Subsystem". In addition to stabilizing the orbiter, the booms will also carry the two magnetometer sensors. One sensor will be placed at the end of each of the booms. This will serve to isolate the instruments from the magnetic field created by the satellite itself. However, since one of the booms and, therefore, the magnetometer sensor, are continuously between the orbiter and the planet, any radio communications will interfere with the accurate sensing of magnetic waves. This interference can be eliminated through the use of a low pass filter in each of the sensors. This device will filter out the high frequency radio waves while allowing the low frequency magnetic waves to pass through the sensors. A low pass filter also eliminates the necessity for the magnetometer booms to be spun as proposed in the preliminary design.

The gravity gradient boom creates an additional problem. It is itself an obstacle between the instruments on the orbiter and the object of their measurements and may, therefore, be read as a rather striking feature on the surface of the planet. This, however, can be easily corrected via preflight programming of the instruments.

Table 2.7 is a summary of the instruments chosen to be carried on the MARTIAN science orbiter. Pertinent parameters for each of the instruments are also given. The instruments are all oriented along the gravity vector pointing toward the planet with the exception of the dust detector which is aimed in the direction of motion of the orbiter. The placement of these instruments on the satellite is shown in Figure 2.1.

Table 2.7

Instrument/previous or proposed mission	Mass(kg)/Power(W)	Data Rate(kbps)
Gamma Ray Spectrometer/ Mariner Mark II	14/5	1.30
NIMS/Galileo	18/8	3.00
UV Photometer/Pioneers 10 & 11	5/6	.50
Dust Detector/Galileo	5/5	.04
Radio Science/(various)	5/18	1.00
Surface Radar Mapper/ Pioneer Venus	9.7/18	.12
Atmospheric Sounder/ Pioneer Venus	10/10	1.00
UV Spectrometer/ Mariner Mark II	3.1/1.7	2.50
Magnetometer/Galileo	6/6	.10



1. Gamma Ray Spec.
2. NIMS
3. UV Photometer
4. Dust Detector
5. Radio Science
6. Radar Mapper
7. Atmospheric Sounder
8. UV Spec.

Figure 2.1: Instrument Placement

Figure 2.2 shows schematically the minimum, peak and intermediate phases of operation of the science instruments and the power requirements corresponding to these phases. The most important impact of the satellite trajectory on the scientific mission of the MARTIAN project is the precession of the satellite orbit. Due to the high eccentricity of the elliptic orbit, instruments such as the dust detector and UV photometer are unable to obtain meaningful data during a considerable portion of the orbital period due to the great distances. Peak operation, therefore, occurs near periapse passage. In a stationary orbit, this would yield great accuracy and detail of information pertaining to the suborbital surface near the periapse point but very little knowledge about the rest of the planet. In our design, however, the precessional character of the orbit means that the requirements for "global" information will be met.

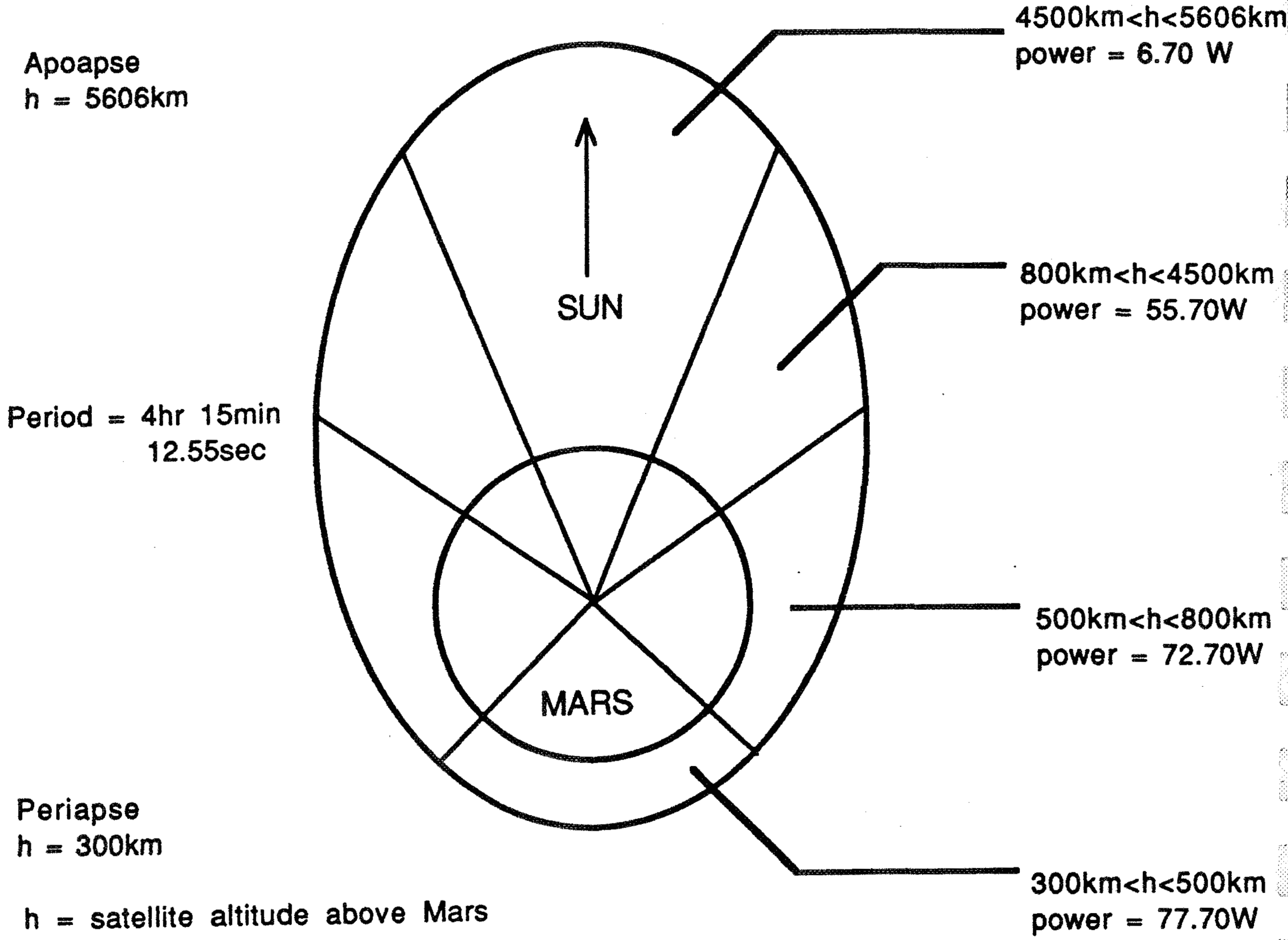


Figure 2.2: Phases of Operation

In the development of the MARTIAN science orbiter, systems integration plays a crucial role. Figure 2.3 depicts the interactions between science and other subsystems contributing to the development of a satellite which would fulfill the requirements outlined above.

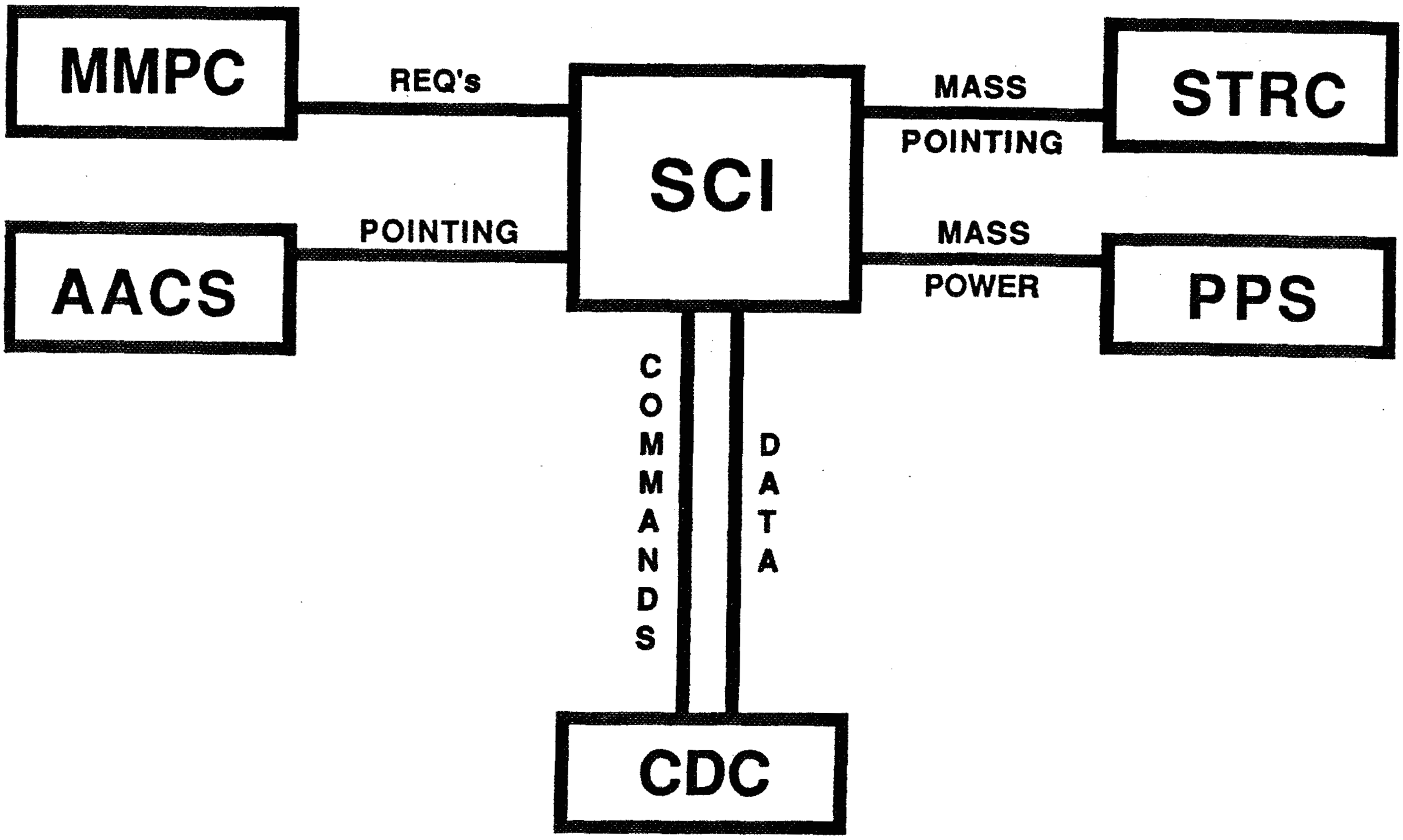


Figure 2.3: Subsystem Integration

A possible enhancement for this or future missions would be the addition of an instrument designed to measure and determine the effects of the solar wind at Mars. To this end, a solar wind plasma analyzer may be added to the science payload to examine properties such as the velocity, density, flow direction and temperature of the solar wind and its interactions with the planet's upper atmosphere. This information would be useful in gaining an understanding of this dynamic force in our solar system as well as an insight into the effects of the solar wind on weather processes on Mars and our own planet.

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Command and Data Control

As a whole the Command and Data Control subsystem will consists of a computer, radio, and an antenna. The CDC subsystem requirements are:

•Collect/send telemetry from all subsystems

•Send/receive commands to all subsystems

•Control power switching for all subsystems

•Accept/carry out commands from mission control

•Receive power from PPS

CDC: Computers

For this mission, we will be using two computers. The first computer will reside in the satellite and will be responsible for all computing needs for the flight out to Mars. It will also, once separation of the satellite from the reentry vehicle has occured, be the only computer aboard the satellite. The second much smaller and more dedicated computer will reside and remain in the reentry vehicle once separation has occured. This second computer will be responsible for the management of the aerobrake maneuver after separation, circularization of the reentry vehicle orbit, and any computing that might need to be performed during the decent of the cargo to the surface of Mars.

Both computers will make full use of the state of the art in both hardware and software, including advances in Artificial Intelligence, while keeping in mind the 1998 technology cut-off. They must also be reprogramable during the mission in case: mission objectives change, to de-bug system once mission has started, uploading of new algorithms, and finally to form a software patch for certain unexpected hardware failures. Both computers must also be able to send/receive commands from all subsytems and accept/process power request from all subsystems. Finally, we will require the main computer, the computer on the satellite, to be able to perform encoding/decoding along with error detection of any sent/received telemetry. We will also have this computer perform data compression of any very high data rate telemetry.

The peak data rate as supplied by the science subsystem analyst is 9.56kbps. The main computer should at the very minimum have sufficient memory to store at least one complete orbit's science and spacecraft health data before downloading the data to the Martain base.

CDC: Radio and Antenna

The CDC subsystem will contain two separate radio/antenna packages. Like the computers a smaller radio/antenna unit will stay with the reentry vehicle after separation for on orbit communications. While a larger radio/antenna package, capable of communications on an interplanetary scale, will remain on board the satellite after separation has taken place. Both units will broadcast on the X-Band frequency (see fig. 3-1). A separate low-gain radio/antenna package for base or plane to satellite communications has been discounted because this extra system would add extra weight and cost to the satellite unnecessarily. It is my belief that the X-Band communications system can meet all communications requirements by itself. Specific requirements for the radio/antenna system are:

- •Sufficient power to send/receive communications for 4yrs.
- •Provide up/downlink for plane to base communications
- •Able to receive/accept commands from mission control
- •Able to send science and other data down to base

RADIO FREQUENCIES

TYPE	ADVANTAGES	DISADVANTAGES
S-Band 2.3 GHz	Being phased out by DSN for deep space communications	
X-Band 8.4 GHz	Required for all 1990+ Deep space comm.	
Ka-Band 32 GHz	Less power required for transmitter Better SNR	No testing in outer space before 1998
LADAR Optical	Very high data rates 100x X-Band	Tech. not ready before 2000+

Option Selected

Fig 3.1

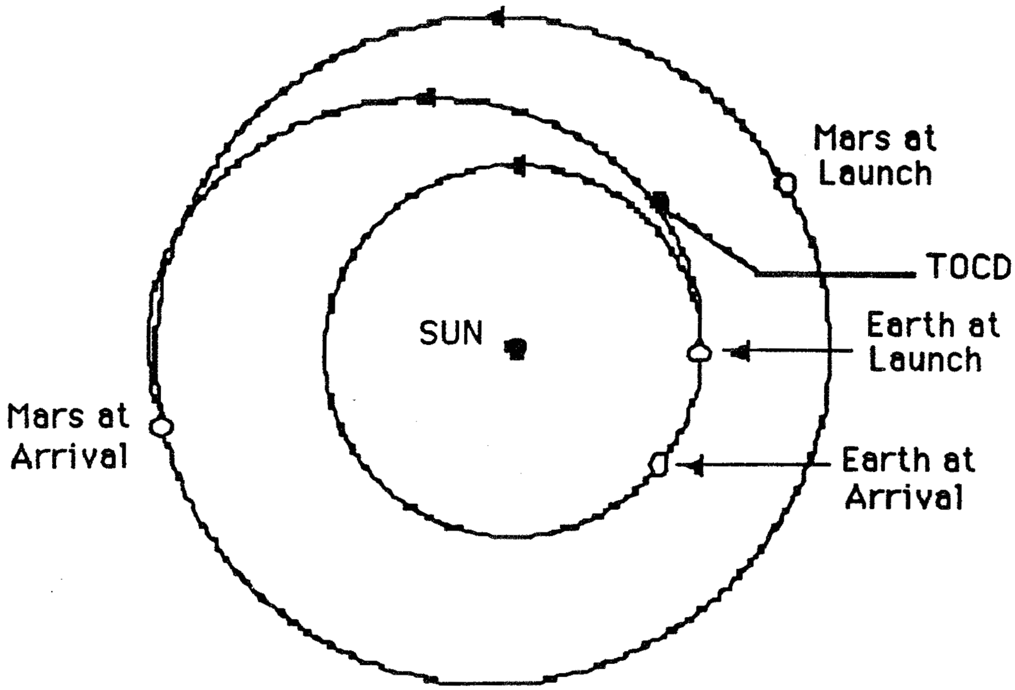
The smaller radio/antenna unit will serve the purpose of receiving commands from the satellite or Mars base such as: when to fire retro-rockets for remaining aerobrake maneuver and landing on the surface. It will also have to be able to send back messages of compliance to these commands and reentry vehicle health. This smaller radio/antenna system will use a 10m whip antenna to allow for lower cost over a parabolic antenna and hopefully better line of sight visibility for the antenna. (The aerobrake shield might occlued a smaller fixed antenna, but the whip antenna should be able to "see around" the areobrake shield.) The reentry vehicle radio/antenna system will only need enough power to perform routine surface to orbit or orbit to orbit communications. It will not be capable of performing interplanetary communications.

The larger of the two radio/antenna units will reside on board the satellite. This system will be responsible for all interplanetary communications and it will be sized in this regard. It will consist of a radio transmitter of P_t=25W matched with a fixed, non-furlable, antenna of d_t=3.7m. (Both of these are nominal values for this type of deep space communications equipment; therefore, these items could be purchased off-the-shelf thus reducing cost and increasing reliability.) For purposes of this preliminary design study, it will be assumed that an established and well maintained line of sight between the two communicating bodies is sufficient for purposes of pointing accuracy.

It will be assumed that the Mars base has radio equipment with sufficient power to communicate with the Deep Space Network. We will make use of this assuption by switching from an Earth based mission control to a Mars based mission control at the point when the spacecraft is equidistant from the Earth and Mars. This has the rather significant impact of reducing the one-way maximum communications distances and therefore the round trip communications times. At arrival the communications distance would be 2.224a.u., but with this maneuver the maximum communications distance will be only 0.4a.u. (distances provided by MMPC). This translates into two-way communication times of 36.94min. and 6.65min. respectively. This savings in time will enhance mission reliability by letting mission control have faster execution of any commands. This Turning Of Communications Destinations, TOCD, also proves quite valuable when the spacecraft is in the critical aerobraking maneuver. Monitoring of the spacecraft will not be from Earth at a distance of 2.224a.u. but from the *surface* of Mars itself thus allowing nearly instantaneous monitoring and control over the spacecraft (see fig 3.2).

The equivalent noise temperature for the downlink of the X-Band frequency is $T=28.5^{O}K$. This will be used as a worse case, because of the lack of water vapor in the Martain atmosphere, Mars will have a much lower thermal noise temperature than the $28.5^{O}K$. Through my research I have learned that at the X-Band frequency attenuation of radio waves by Martain dust storms is not a significant factor for CDC sizing purposes.

Mission Geometry



Turning Of Communications Destinations

	TOCD	Non-TOCD
Max comm. distance	.4 a.u.	2.224 a.u.
Two-way comm. times	6.65 min.	36.94 min.
Aerobrake comm. times	.99 sec.	36.94 min.

fig 3.2

Radio/antenna sizing

X-Band

Uplink	Downlink
f=7170 MHz	f=8450 MHz
l=.0418 m	l=.0355 m

l=wavelength

P_t=25 W

d_t= 3.7 m

d_r=70m

Z=.55

D=.4 a.u.

C=2.998x10m/s

$L_p = 10\log_{10}((4\pi D)/l)^2 = 133.26$ dB

$G_t = G_r = 10\log_{10}((4\pi A_{eff})/l^2) = 35.8$ dB

$P_r = P_t((fzd_r d_t \pi)/(4CD) = 5\times10^{-14}$ W

Minimum P_r that DSN can receive is 4×10^{-21} W so we are assured that our communications signal will be received.

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AACS

Introduction:

Requirements:

The attitude and articulation control subsystem consists of sensors, thrusters, and pointing/deployment actuators controlled and coordinated in such a way as to meet and/or exceed the stated requirements, described below.

The applicable requirements distilled from the RFP are:

- optimization of performance, weight, and cost
- reliability and ease of operation
- off-the-shelf, pre-1998 hardware
- several missions possible
- four year lifetime

The requirements given for AACS specifically are:

- send telemetry to CDC
- accept commands from CDC
- power from PPS
- outputs
  - spacecraft attitude
  - instrument pointing
  - antenna pointing
- inputs
  - attitude references
  - actuator positions
  - thruster valve actuation

Method of Attack Identified:

Figure 4.1 shows the method of attack followed in meeting the stated requirements; it is self explanatory.

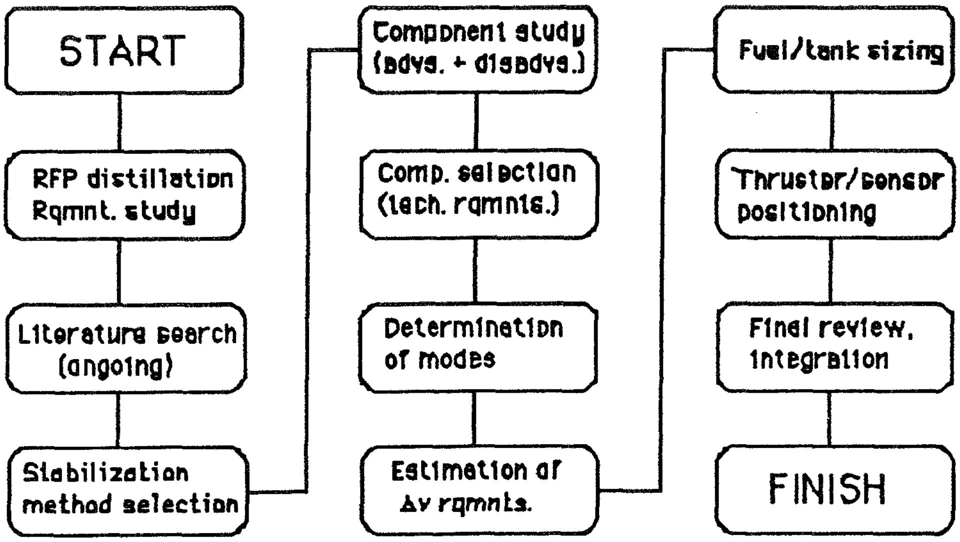

Figure 1: Method of Attack

General Control Philosophy:

In order to demonstrate the general control philosophy followed, a block diagram is presented as Figure 4.2.

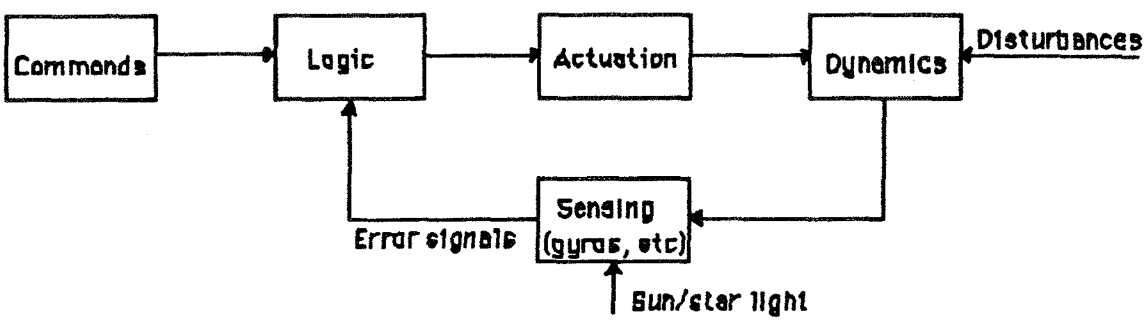

Figure 2: General Control Philosophy

Commands are generated indicating a desired positioning, and computer logic determines the torque needed in response. Thrusters provide that torque, causing a new attitude to be achieved as the spacecraft moves. This position, taking into account any disturbances encountered, is sensed, completing the loop.

Stabilization Concerns:

One of the first decisions to be made in the design of the subsystem is the method of stabilization. Simple spin, dual spin, and three axis control were compared, with passive control by gravity gradient also examined.

The simple spin method was rejected immediately because of science instrument requirements; a number of instruments must have a fixed orientation. Dual spin was attractive because of its scanning and inertial pointing capabilities, low disturbance sensitivity and minor principle axis stabilization, but was rejected due to its very complex dynamics, the high cost of precision pointing, sensitivity to mass properties and mass balance compensation requirements. Three axis control, with its high precision and maneuverability, adaptability and high accuracy was chosen as the base stabilization philosophy.

Choosing three axis control then required another decision to be made, between momentum exchange and mass expulsion methods. Momentum exchange was rejected due to rotational and momentum dumping complexities, nutation, and precession. Mass expulsion was chosen for its quick response, high accuracy, and inertial orientation.

The disadvantages of the chosen methods as well as the advantages of the rejected methods do cause concern. Maintaining the desired orientation with an all thruster approach is very expensive in terms of propellant required, and limits spacecraft lifetime. Propellant could be conserved by a momentum exchange approach, for example. Power requirements and complex thrust vector control are other drawbacks to the chosen approach.

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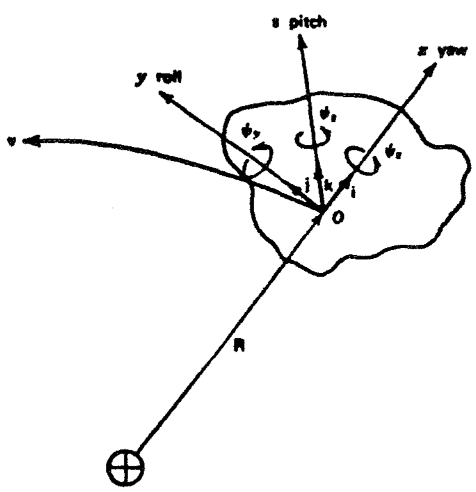
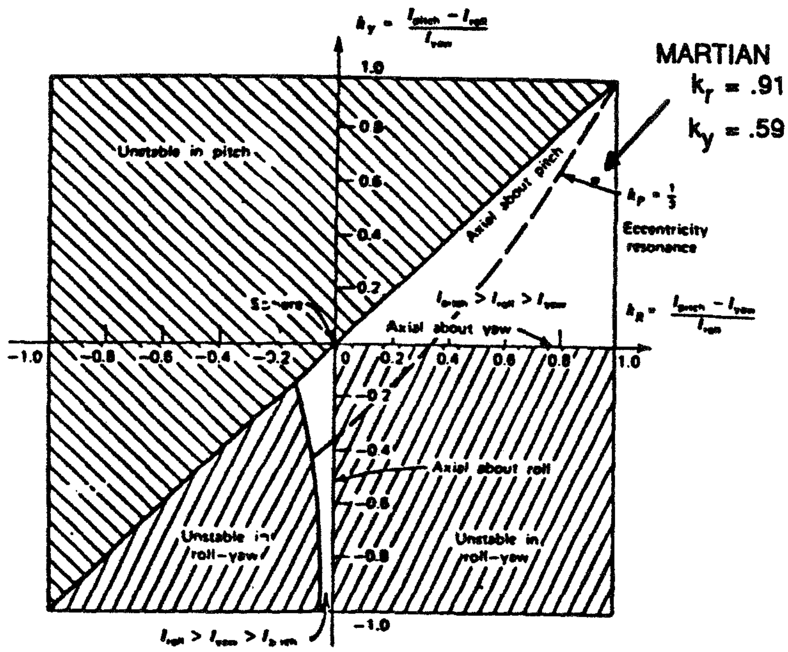
For these reasons, the MARTIAN satellite will employ passive gravity gradient stabilization in addition to three axis. Two 7 1/2 meter booms will be deployed when the desired orbit is reached. The gravitational field difference between the end of the booms and the main spacecraft body causes a restoring torque, aligning the booms with the local vertical axis.

Given the attitude geometry of Figure 4.3, the gravity restoring torque is given by $G = (3\mu)\,(I_{xy}^2 + I_{xz}^2)^{1/2} / R^3$ (Kaplan,1976,p.201) where I_{xz} and I_{xy} may be approximated by $I_{xz} = I_x\phi_y + I_z\phi_y$ and $I_{xy} = I_x\phi_z + I_y\phi_z$. R is the orbit radius and $\mu = 4.305 \times 10^4$ km^3/sec^2 at Mars. The magnitudes of the torques thus calculated for various orbit altitudes are very similar to those of proven satellites.

Figure 4.4 shows that for gravity gradient stability, $I_{pitch} > I_{roll} > I_{yaw}$. For the MARTIAN satellite, $I_{pitch} = 958.2$, $I_{roll} = 850.8$, and $I_{yaw} = 181.0$, meeting this requirement.





Figures 4.3 and 4.4: Attitude geometry and Grav. Grad. Stability Region (Kaplan,1976,p.202,4)

Gravity gradient stabilization offers many advantages. A great burden is taken off the active thruster system, because the passive control provides the major stabilization. Thrusters do the fine tuning as well as damp out oscillations. Additionally, science instrument orientation towards the planet is not a problem. The magnetometer will be mounted on the end of the gravity gradient boom, and the other instruments mounted on the main spacecraft body in their required aiming orientations (see Positioning and Pointing).

This approach, then, eliminates the need for a scan platform and all the associated complexities. Finally, thruster off failure is no longer disastrous. In the event of multiple thruster failure (which is very unlikely), the spacecraft will not immediately tumble, due to the gravity gradient boom. More time is thus available to effect necessary repairs, if possible.

Component Study and Selection:

In this section the need for specific instrumentation and the rationale behind its selection is presented.

Accelerometers: The accelerometers are part of the sensing block in the control diagram above (Figure 4.2). It is highly desirable to implement fiber optic accelerometer technology for the reasons described in the gyroscope section below. One accelerometer per axis in both the aerobrake vehicle and the orbiting satellite is necessary, each having an approximate mass of 1 kg and approximate power of 4 W.

Actuators: Rotation of the ultraviolet spectrometer, deployment of the gravity gradient boom and antenna, and actuation of the solar arrays are the four areas considered. Simplicity and reliability are essential here, because a frozen antenna or an undeployable boom would bring severe problems to the mission.

A very simple gravity gradient boom deployment scheme was used on

UOSAT in the early 1980's: a half inch diameter beryllium copper tube, unwinding from a drum (Hodgart,1982,p.380). This design has been chosen because of its simplicity, utility and previous successes. Estimated deployment motor mass is .75 kg with a peak power of 10 W.

The satellite antenna will be placed on a simple two axis antenna drive system providing large angular deflections. No detailed selection process is described here; many of the flight proven devices available will meet the requirement. We assume such a system for the satellite with approximate mass 7 kg and approximate power 20 W. No such system is required on the aerobrake, according to CDC.

The ultraviolet spectrometer must have the ability to point both at the planet surface and out to space, for calibration purposes. This will be accomplished by a simple rotational actuator, rotating the instrument from its star oriented rest position, 90° to face the planet surface. Estimated mass is .75 kg with a peak power of 5 W.

Finally, a standard flight proven solar array drive with a torque capability of at least 50 N-m is chosen for array manipulation. Estimated mass is 4 kg with peak power 9 W. Four such actuators will be necessary, two each for the satellite and aerobrake. Once again, it is assumed that one of the many available flight proven actuators will perform well; no specific choice is made.

Computer: To promote simplicity and cost effectiveness, the AACS control logic will be handled by the main CDC computer. Sensor telemetry will be sent and resulting commands received from the device, as depicted in the control block diagram (Figure 4.2).

Gyroscopes: To provide essential attitude determination information, two gyroscopes are oriented in a "V" formation to sense spacecraft movement in the three orthogonal directions. A third gyro will be provided for redundancy. This three gyro system will be employed in both the aerobrake and the satellite.

Recent advances in fiber optic technology render trade studies on non-fiber devices obsolete. When the requirements concerning reliability, lifetime, performance, etc. listed earlier are considered, it becomes obvious that fiber devices are the ones to choose. They are small, rugged, virtually maintenance free, without complicated mounting systems, require no warm up time and are even low in cost (IEEE,1986,p.54).

The Fiber Optic Rotation Sensor is especially attractive (Draper, 1984,p.14), with an estimated life of ten years, ten kg mass, and less than 10 W of power required, and is chosen.

Star Tracker: To determine the location of MARTIAN in space, the satellite is equipped with a star tracker detecting stars and their locations in relation to the vehicle. Additionally, gyro drift rates, various misalignments, and the absence of precise initial attitude reference necessitate star tracker use.

In choosing a star tracker, solid state imaging is considered much more desirable than tube-type devices, with their high voltage requirements, error susceptibility and magnetic field problems (Armstrong,1985,p.342). Also, a Charge-Coupled Device (CCD) is desired, allowing tracking to occur even with considerable background light.

Three such solid state devices exist and are available: JPL's ASTROS, TRW's MADAN, and Ball Aerospace's RFT. The choice is quickly made, because the MADAN is designed primarily for Earth-orbiting spacecraft and the RFT uses a Charge Injection Device, with predicted performance characteristics inferior to CCD's. The ASTROS, which has a CCD, is chosen. It is assumed that the device can be modified or improved to suit the MARTIAN application specifically, as the "planetary ASTROS" was developed for the Mariner Mark II program. It has a projected mass of 8 kg and power consumption of 11 W.

Sun Sensors: The sun sensor determines the sun's position relative to the satellite, and is generally necessary for calibrated rotational motion about specific axes. Two sensors are placed on the spacecraft in order to function in the desired manner.

The two sun sensors may be chosen from the many flight-tested models available, with mass approximately 1 kg and an approximate 2 W power requirement.

Table 4.1 summarizes the components chosen, their masses, and power requirements.

Device (#)		Power (W)	Mass (kg)
Accelerometer	(6)	24	6
Actuators			
-Antenna	(1)	20	7
-Boom	(1)	15	2
-Solar array	(4)	36	16
-UV Spect.	(1)	5	1
Gyroscope	(6)	60	60
Star tracker	(1)	11	8
Sun sensor	(2)	2	4
	TOTALS	173	104

Table 4.1: Comp. powers and masses

The obvious overlap of instrument functions (e.g. gyros, star tracking, and the sun sensors can all provide yaw error information) greatly increases the reliability of the AACS. Safety is assured in the event of component problems/failure.

Positioning and Pointing:

In this section the rationale behind the placing and aiming of thrusters, sensors, and other instrumentation is presented. These general guidelines were given to STR, which did the actual specific positioning. Reference is made to the spacecraft diagram in the STR subsystem report.

Firstly, a number of the motion sensing devices may be buried within the structure of the spacecraft, such as accelerometers, the computer system, and the various gyroscopes. Three gyroscopes and three accelerometers were placed on both the aerobrake and satellite for optimum sensing with redundancy.

The equipment used to locate the satellite in space are the two sun sensors and the ASTROS star tracker. Their positioning was fairly simple. In order to maximize exposure time to the sun, the sensors are placed on the top of the spacecraft. The star tracker, in order to remain fixed on its three star base both in orbit and enroute, is mounted on side of the satellite, pointing parallel to the arrays, near the antenna. In orbit it aims perpendicular to the local veritcal, guaranteeing once again the optimum view.

The two axis antenna actuator and antenna placement on the side of the satellite guarantee all necessary maneuverability. The ultraviolet spectrometer 90^{o} actuation device allows the instrument its required views of both the planet surface and its calibration stars. The other instruments were positioned by the SCI and STR subsystems, with very little fine tuning or pointing necessary.

The selection of AACS thrusters was performed by the PPS subsystem, and their placement was determined by AACS and STR. As much translational and rotational control along and about each axis as possible was desired for both the aerobrake and satellite. To apply as true a torque as possible, each thruster per firing pair was placed an equal distance from the center of mass.

The satellite utilizes 12 thrusters, enabling six degrees of freedom. Moment arms for each thruster pair have been maximized for minimum propellant use. Because the aerobrake blocks out one side of the entry vehicle, it carries eight thrusters, once again each with the largest possible moment arm. Control maneuvers will, if possible, be carried out on the minimum moment of inertia axes.

Delta-V Estimation and Tank Sizing:

It is impossible to predict exactly how much propellant will be required for the attitude control, repositioning, and other functions of space vehicles. For MARTIAN, estimation of these requirements was completed using various equations, and large safety factors were built in to assure that fuel requirements would be met throughout the spacecraft's lifetime.

For a pair of thrusters separated by a distance l and with angular moment of inertia M, it is known that $(F)(l) = (M)(a)$, where a is the angular acceleration. Now $a = (2)(ø)/t^2$, where ø is the displacement angle and t is the elapsed time, so $F = (2)(M)(ø)/t^2 l$. Because $F = (w)(I_{sp})$ where w is weight flow in lb/sec, we can see

$$M_p = (2)(M)(ø)/(I_{sp})(t)(l)(g_o)$$

where M_p is the propellant mass used, $g_o = 9.81$ m/sec^2, and $I_{sp} = 300$ sec for the thruster system chosen.

For enroute attitude control propellant estimation, it is assumed one rolling rotational maneuver will be performed every day for the 326 days of the flight. Given the approximate moment 88200 kg-m^2, an l of 22 m, and taking a time of 600 seconds, the required propellant mass is (326)(.029) = 9.45 kg. By the same method, the yaw axis maneuvering estimated propellant mass is (10)(.038) = .38 kg, assuming ten turns.

After the satellite is separated from the aerobrake by pyrotechnic bolts, it must accelerate away from the aerobrake. Assuming a linear increase in acceleration to 2 m/s^2 in 15 seconds, the total ΔV required to move away and then stop once again is 30 m/s. Using the rocket equation, the estimated propellant mass to separate is 4 kg.

After separation, the satellite must have a lifetime of at least four years. Due to the gravity gradient stabilization technique described earlier, fine tuning propellant mass is minimized. Based on the AACS estimated fuel requirements for similar spacecraft (Chubb,1975,p.226), the MARTIAN satellite has 20 kg of propellant allocated for this purpose. Also, 28 kg of fuel has been provided for satellite repositioning (providing a ΔV capability of approximately 235 m/s), and 5 kg for control during earth orbit escape.

Finally, after separation the aerobrake must execute various attitude control turns. For one rotational maneuver about the yaw axis, approximately .3 kg of fuel is required. For safety, 5 kg of AACS fuel will be prescribed for that purpose.

Table 4.2 summarizes the AACS fuel requirements.

To maximize rotational torques, the burns occurring between Earth and Mars will be performed by the aerobrake thrusters. As can be seen, the total propellant masses are 52.0 kg and 19.8 kg on the satellite and aerobrake, respectively. Assuming the density of hydrazine to be 1060 kg/m^3, tank volumes of 49057 cm^3 and 18679 cm^3 are required. STR has determined the shape and placement of these central tanks on both vehicles.

Event	Vehicle	M_p (kg)	Tank	Impulse (kN-s)
Earth escape	A/S	5	A	14.7
Mars transfer	A/S-roll -heading	9.45 .38	A A	27.8 1.1
Separation	Sat	4	S	11.8
Sat. A/C	Sat	20	S	58.9
Sat. repos.	Sat	28	S	82.4
Aerob. control	Aero	5	A	14.7
	TOTALS	71.8		211.4

Table 4.2: AACS fuel requirements

Control Modes:

AACS has been broken down into five specific control modes, with different functions and instruments in each.

In the acquisition mode, the spacecraft will get a fix on its position relative to various heavenly bodies. Onboard star catalogs will provide the celestial reference for the tracker, and the sun sensor will determine the relative positions of the earth, sun, and spacecraft. With this foundation, the other sensing instruments can perform their tasks.

In the cruise mode, the two major functions are navigational checking and spacecraft health updating. Unforseen perturbations such as solar torques, magnetic fields, internal accelerations, etc. cause changes in spacecraft dynamics. These will be registered by the gyroscopes and accelerometers, telemetry sent to CDC, and correcting torques applied. Additionally, software will make periodic checks of all AACS devices to ensure proper functioning. This is especially applicable to thrusters.

The maneuvering mode encompasses any necessary rotational and small translational movements. Commands sent by CDC activate thrusters in the required combinations to effect the desired motion. During the flight to Mars, turns will be performed to initiate the aerobraking process, and once on orbit, various turns and repositioning may be needed. Additionally, boom deployment will occur in this mode.

In the fine point mode, actuation devices will combine with thrusters (if necessary) to set the antenna and science instruments in the proper configurations. Once again the CDC computer will direct this activity.

Finally, should any more repositioning be necessary, the Delta-V mode will be implemented, providing a fine tuning to the booster engines present. Separation of aerobrake and satellite occurs here.

Integration:

A description of interactions between AACS and the other subsystems is presented here. Figure 4.5 summarizes this section.

The power needed to run the AACS devices is provided by PPS. Average and peak power requirements were calculated and provided to PPS. PPS also handled thruster selection and the fuel/tanking to operate them, sizing estimates having been provided by AACS.

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Spacecraft positioning immediately preceding and during the aerobraking maneuver is handled by AACS and requires input from AERO. Post separation control is another concern; AERO providing a general idea of the degree of control necessary.

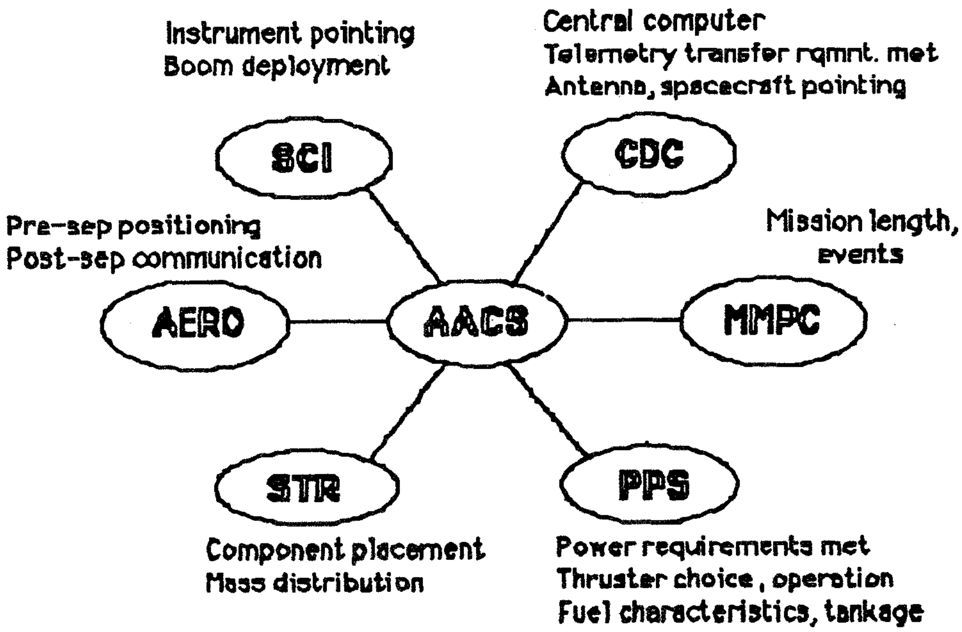

Figure 4.5: Integration Activities

CDC operates the computer which commands AACS modes, and obviously there is much telemetry transfer between the two subsystems. Antenna and valve actuation and spacecraft positioning for communication purposes represent further interactions.

SCI provided AACS with magnetometer boom specifications as well as information about instrument pointing. It was also determined that pointing requirements were such that no scan platform was necessary, instruments being mounted directly on the spacecraft body.

MMPC provided length of mission data and general control consultation.

Finally, without a structure, AACS would have nothing on which to place its various instrumentation. Suggestions concerning optimal component placement were provided to STR, which then calculated final moment of inertia values. Component masses and dimensions were provided by AACS.

Problem Areas and Future Concerns:

There are a few problem areas requiring future attention. The first of these concerns the thruster control of the aerobrake. Because the aerobrake shield prevents the aiming of thrusters to the front, full rotational and translational control is not provided. Although pure roll of the entry vehicle is possible, turning about the pitch and yaw axes is not possible without some translational perturbation.

Secondly, because of the large dimension and mass of the aircraft, the spacecraft is necessarily also fairly massive and large. If for any reason rapid movement is required, a great quantity of propellant must be consumed. It is hoped that whatever enroute movement is necessary will be quite gradual.

Bending interactions between the thrusters and the spacecraft structure should be more thoroughly investigated. Repeated use of a particular thruster may induce a high cycle fatigue on the spacecraft that would have to be compensated for, either by material reinforcement or a more varying thruster usage schedule.

Finally, hydrazine thrusters are obviously limiting in that when the propellant is gone, the thruster is useless and the spacecraft will eventually be lost. An area for future study therefore is an on-orbit thruster refueling scheme, replenishing the valuable hydrazine and extending spacecraft lifetime considerably.

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Aerobraking, Reentry, and Landing Subsystems

The last portion of the Martian mission consists of capturing a Martian orbit, aerobraking to a desired altitude, deploying the satellite, entering the Martian atmosphere, and landing at a suitable ground site. These responsibilities have been grouped together in to the AERO subtask. This part of the project report will describe the components used to carry out these four tasks. The method of analysis and the trade studies performed will also be presented along with the process of interactions that was necessary between the AERO group and the other subsystem groups.

Requirements

The general requirements that apply to the overall mission,such as keeping the cost low and keeping the design simple, also apply to the AERO subtask, but there are several requirements that are specific to the aeorbraking, reentry, and landing portions of the mission. The most important of these is to use aerobraking techniques to reduce the fuel needed to achieve the desired Martian orbit. This also implies that we must find a way of analyzing aerobraking parameters which in this case is the AEROB computer program. A second important requirement is that we separate the satellite from the reentry vehicle and install it into the proper elliptical orbit. This requires the development of a separation technique and a method for deploying the satellite at the proper altitude. The reentry vehicle must be able to sensor inputs (attitude, temperature, pressure, payload status, etc.) and communicate these to the satellite along with receiving commands from the satellite computer (see CDC subgroup). The reentry vehicle must be able to decelerate itself upon reentry and to withstand the resulting stresses. This requires thermal shielding, parachutes, retro systems, and shock absorbing payload mounts. The reentry vehicle must be self-powered (see PPS) . The reentry vehicle must have a means of control and stabalization (see AACS). The payload must be protected against extreme temperatures, pressures, radiation, and stresses in all phases.

Subsystem Interaction

In order to produce a reasonable design concept, there are on-going interactions between the AERO subsystem and the other subsystems (Fig. 1). The AERO subgroup works most closely with structures (STR) and power and propulsion (PPS). AERO must give STR the constraints on spacecraft configuration due to wake considerations along with the AERO component weights and sizes. STR must inform AERO of the overall spacecraft weights and of the type of support structure used in the reentry vehicle. AERO and PPS must work together to determine engine weights, I_{sp} values, velocity changes needed, and potential savings over an all propulsive case. Mission planning (MMPC) must supply the AERO subgroup with the expected hyperbolic approach velocity and the desired orbit of the satellite. Command, data, and control (CDC) is responsible for determining the type of communications gear, sensors, and computer that is needed on the reentry vehicle. Attitude, articulation, and control (AACS) is responsible for developing a three-axis thruster system that will keep the vehicle stable and under control during aerobraking and upon reentry. There is little correspondence between science (SCI) and Aero, since data will be primarily sampled after separation of the reentry vehicle and the satellite.

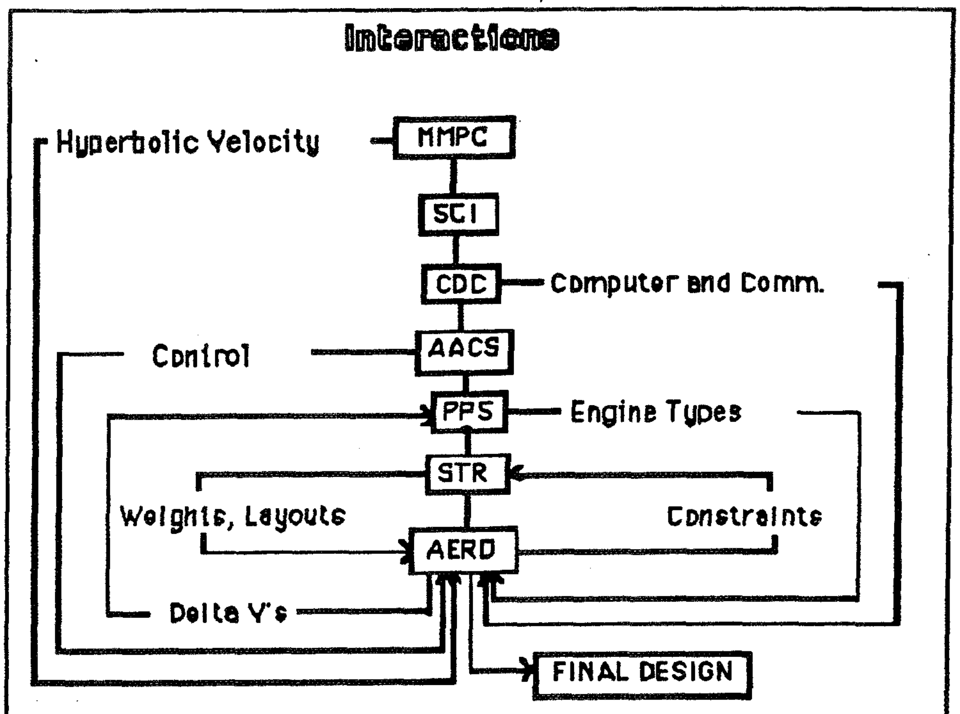


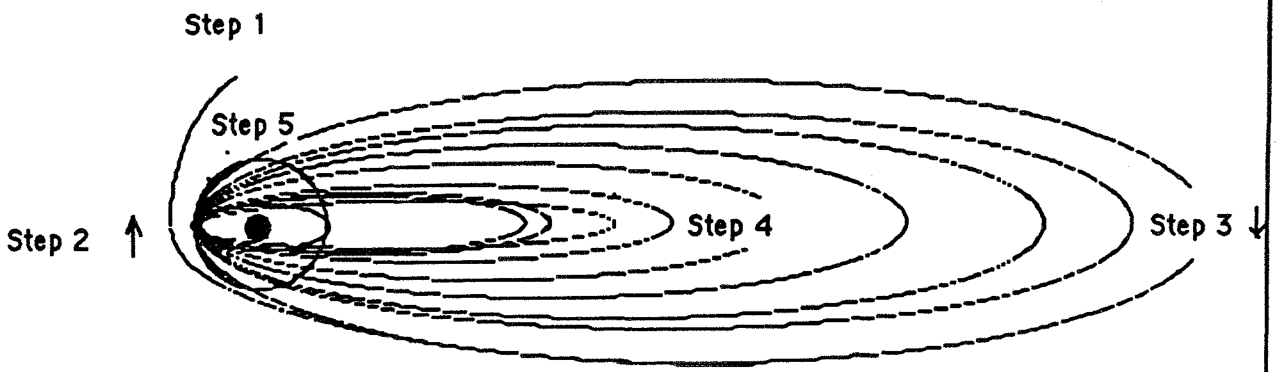
Fig. 5.1

Aerobraking

Now that the interaction process has been described, the individual sequences will be presented along with the related components and the method of choosing these components. The first of these is the aerobraking procedure. Figure 2 roughly outlines the aerobraking sequence including deployment of the satellite. The spacecraft approaches Mars at a velocity of 2.464 km/sec (provided by MMPC). The spacecraft must be positioned so that it approaches periapse with the rear engine pointing forward. This allows the engine to provide the -615 m/sec velocity change needed to capture an elliptical Martian orbit. This engine and its tanks are then jettisoned to reduce weight. After this initial burn, the spacecraft is slowly rotated so that the second smaller rear engine can again provide a negative velocity change at apoapse of

-1.06 m/sec. This is also the proper orientation for atmospheric breaking (nose down in the diagram). Through successive passes through the martian atmosphere, the spacecraft utilizes aerodynamic drag to reduce an elliptical orbit with only minor orbital corrections needed at each apoapse (commanded by the satellite). Each pass descends to a periapse altitude of 3724 km, allowing the aerobrake shield to reach a maximum of 600 ^{o}C. The initial semimajor axis was set at150,000 km. When the spacecraft has reached an apoapse altitude of 6347 km, the satellite will be separated from the reentry vehicle to be installed into its proper orbit. The velocity change required by the satellite is 92.45 m/sec with a corresponding fuel mass of 13.08 kg. The reentry vehicle will continue to reduce its orbit until it reaches an apoapse altitude of 3600 km, where it will then circularizes its orbit with a velocity change of 161.7 m/sec. The reentry vehicle will coast in this orbit (up to seven days) until the satellite has determined that the landing site is suitable.

Aerobraking



Step 1) Hyperbolic approach, V = 2.646 km/sec

Step 2) Orbit capture
- altitude = 3600 km
- delta V = -615 m/sec
- spacecraft started rotating to reorient it for step 3

Step 3) Reduce orbit to initiate aerobraking
- delta V = -1.06 m/sec

Step 4) Deploy saatellite at the proper altitude

Step 5) Circularize the reentry vehicle's orbit and then enter the atmosphere when the satellite gives the command

Fig 5.2

Another option investigated was the possibility of separating the spacecraft into three or four separate vehicles prior to the aerobraking sequence with two or three vehicles carrying the payload and another carrying the orbital satellite. This had the advantage of lower shield sizes and lower individual vehicle masses, thereby simplifying parachute and landing analysis and also shield structural and assembly analysis. This option was not chosen due to constraints imposed by the payload. It proved infeasable to break either of our contracted payloads into smaller segments. It was also felt that this method might also reduce our adaptability to future missions.

Most of the previous orbital and aerobraking analysis was accomplished using the AEROB computer program developed by Stephen Hoffman[1] and modified by Eric Johnson of the STR subgroup. The modifications allows the user to deploy the satellite at any apoapse pass. They also enable the program to print out additional information concerning satellite orbital parameters and aerobrake pressures used for structural analysis. Several important input parameters had to be determined before using AEROB. The shield projected area was confined by the size of the payload, since the shield must protect the entire 21m long cylinder that encapsulates the payload. The shield area used was 121 m^2, and the method used to arrive at this is discussed later. The maximum shield temperature was set at 600 ^{o}C, which directly controls how deep the spacecraft can penetrate the atmosphere on each pass. This was found to be the optimum temperature, since at higher temperatures the spacecraft dissipated orbital energy too fast crashing into the planet, and at lower temperatures the spacecraft is not utilizing the full aerobraking potential. Figure 3 illustrates the relationship between the orbit semimajor axis and the final mass using the data listed in table 1. The highest semimajor axis that could reasonably be considered within the influence of Mars was chosen, setting the value at 150,000 km. Engine parameters were provided by PPS. After orbital, engine, and satellite parameters were entered, an initial spacecraft mass was selected that produced a final reentry vehicle mass of approximately 2100 kg. A list of the important input parameters and the corresponding results are provided in table 1.

Simplified Aerobraking Parameters:

Inputs:

Initial spacecraft mass: 4800 kg
Shield projected area: 121 m^2
Coefficient of drag: .8 (ref. 1)
Aerobrake weight factor (compensates for an elliptical surface area): 7.9 kg/m^2
Engine I_{sp} values: 300 sec, 293 sec, 293 sec
Tankage factor for engine 1: .04 (jettisoned)
Tankage factor for engine 2/3: .12 (not jettisoned)
Engine 1 mass: 133 kg
Engine 2/3 mass: 60 kg
Maximum allowable shield temperature: 600 ^{o}C
Satellite mass: 400 kg
Initial periapse radius: 3600 km
Final periapse radius: 3600 km

results:

Number of orbits: 9
Time: 23.21 days
Final spacecraft mass (landing vehicle): 2091.88 kg
Shield mass: 955.9 kg
The satellite was jettisoned on orbit 5 using 13.08 kg of fuel.
Aerobrake pressures, highest: 40.059 Pa
lowest: 23.386 Pa

Note: Original atmospheric densitys given for AEROB earlier in the semester were low by a factor of a hundred. These results reflect the corrected data.

Table 5.1

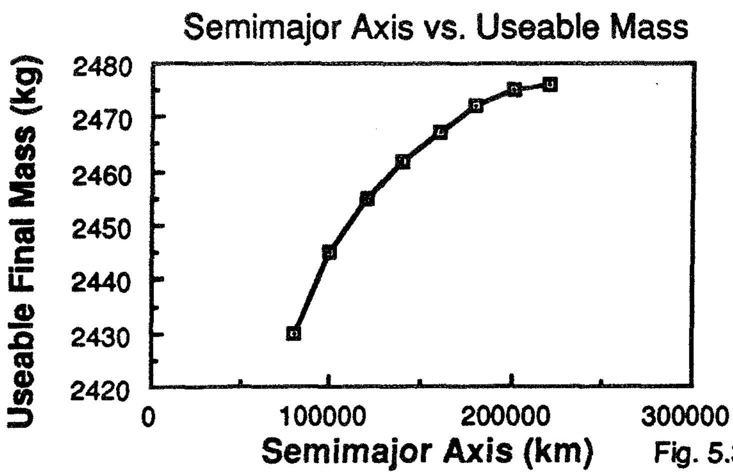


Fig. 5.3

The aerobrake shield is the component that enables the spacecraft to utilize atmospheric drag to reduce its elliptical orbit. The first shield shape considered was a flat circular shield, but this shape was discarded due to the instabilites created by the drag forces on the shield. A hemispherical shield also seemed a logical choice. It provided a more protective wake protection, and it offered greater stability if the spacecraft center of gravity could be located ahead of or sufficiently close to the shield's metacenter. Unfortunately, if the shield was designed large enough to protect the 21m x 6m cargo bay and the satellite, it was simply too massive to reassemble, support, or control. Our choice for a shield shape is an ellipsoid, cut lengthwise (Fig. 4). This shield shape with spacecraft components positioned as in figure 5 still offers sufficient protection and stability while cutting the projected shield surface area from 380 m^2 for the spherical shield to 121 m^2. The elliptical shield's mass was roughly a third of the spherical shield's mass. The materials used in the construction of the aerobrake shield include the SLA-561 silicone ablator skin, chosen for the Viking mission as a minimum weight design; an aluminum back-up shield; an

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asbestos phenolic base for thermal protection; and an aluminum alloy support structure[5]. These materials also allow the shield to withstand reentry temperatures through ablative heat removal.

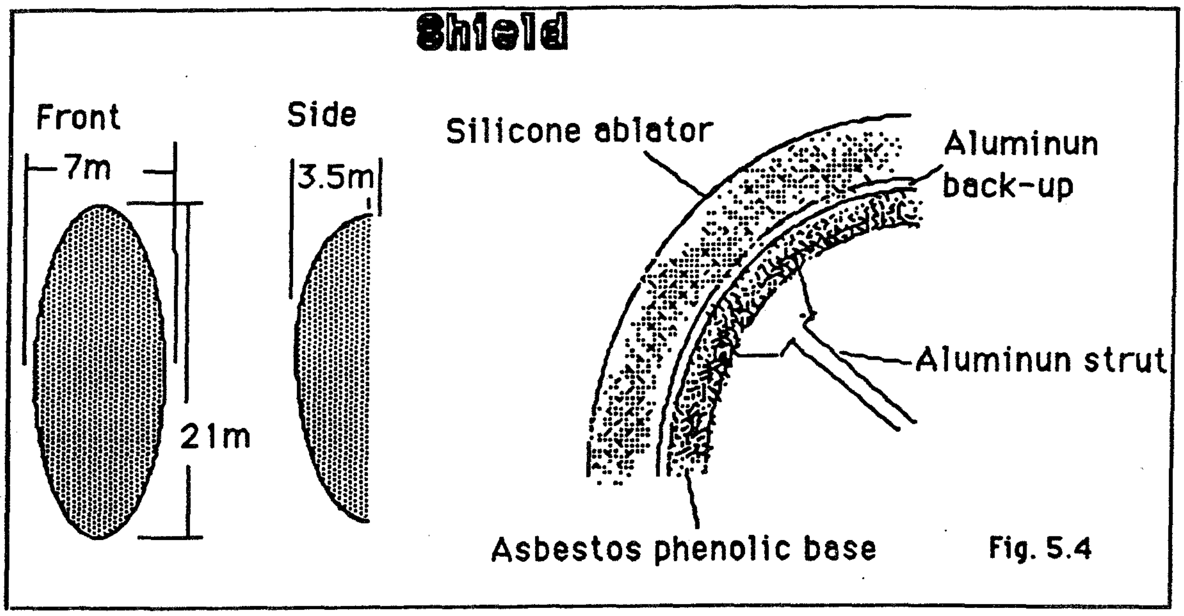

Fig. 5.4

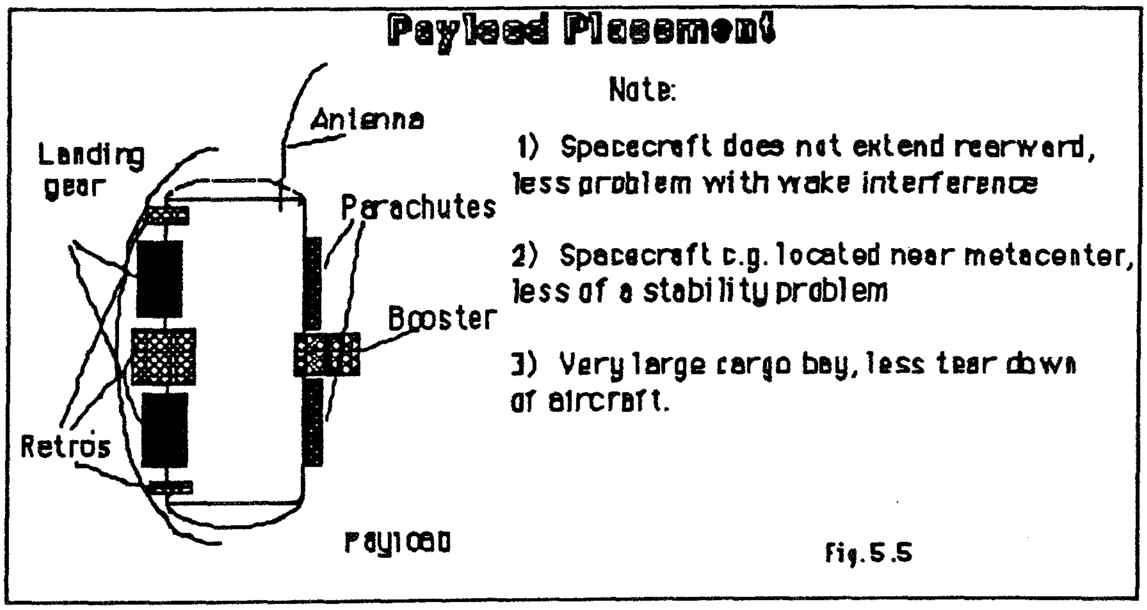

Fig. 5.5

Separation

Although the separation of the satellite and the reentry vehicle occurs within the aerobraking sequence, it is discussed in greater detail in this section. When the spacecraft has aerobraked to the desired apoapse altitude for the satellite, it will be pyrotechnically separated from the reentry vehicle, which will continue to reduce it semimajor axis until it circularizes at 3600 km. Waiting until after spacecraft circularization would require an unnecessary burn to boost the satellite into the higher orbit. This separation of the reentry vehicle and the orbiting satellite will be accomplished using a pyrotechnic system (Fig. 6 a,b,c) similar to that used to separate the shuttle's solid rocket boosters (SRB) forward section from the main external fuel tank. This system employs double-ended, tandem piston separation bolts using NASA Standard Initiator (NSI) pressure charges. These bolts are pyrotechnically actuated with either cartridge sufficing to sever the bolt on a predetermined fracture plane. The bolts can withstand 202,000 lb of tension and 75,700 in-lb of bending. Shock absorbing honeycomb material behind each bolt serves to reduce the shock experienced by the satellite or the reentry vehicle. These bolts are designed to produce minimal debris, thereby reducing the risk of spacecraft contamination or interference[7].

Reentry

After the reentry vehicle has circularized to its predetermined coasting orbit, the satellite will check the available landing sites for atmospheric disturbances or extreme surface irregularities. It has sufficient power to do this for seven days if necessary. When the satellite has determined the suitability of the landing site, the reentry vehicle will be commanded to enter the Martian atmosphere, maintaining a proper angle of attack and flight path angle so that it can reach a

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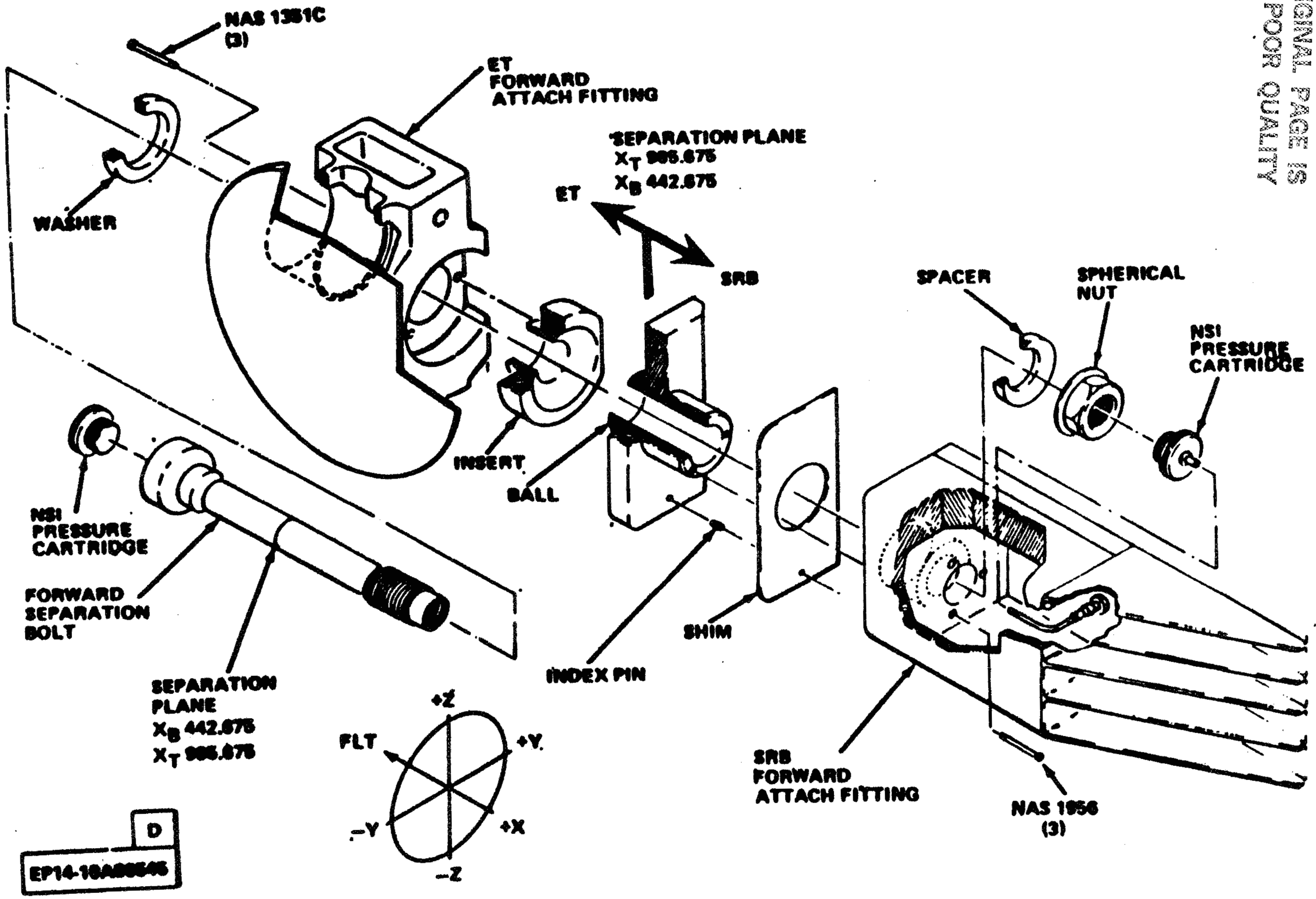

Figure 6-a FORWARD ATTACH FITTING DETAIL

velocity of approximately M2.1 when the dynamic pressure is approximately 8.62 PSF. At these conditions two Viking-type parachutes will be deployed, assisting the separation of the shield and lander. The shield remains on the reentry vehicle until parachute deployment to provide upper atmosphere deceleration, reentry thermal protection (ablation), and a lifting surface for attitude control and further vertical deceleration. The separation of the shield and the lander will be accomplished using the previously mentioned separation bolts located at the shield structure-cargo bay interface.

The parachute system used on this mission borrows heavily from the earlier Viking technology. This allows our mission to employ proven technology and eliminate the cost of developing a new system. The lander will be decelerated from M2.1 to a terminal velocity of approximately 100 m/sec (see PPS) using two 6.2 m disk-gap-band parachutes shown in figure 7. The chutes are deployed by two mortars located on top of the lander. The first two chutes are located sufficiently back of the lander to avoid supersonic wake problems, and the second chutes are located closer to the lander to reduce line tension and to avoid interference between chutes. A system of kevlar briddles will be used to equally distribute the tension along the top of the lander. At a predetermined altitude the parachute system will be released and the landing retros will be engaged[3].

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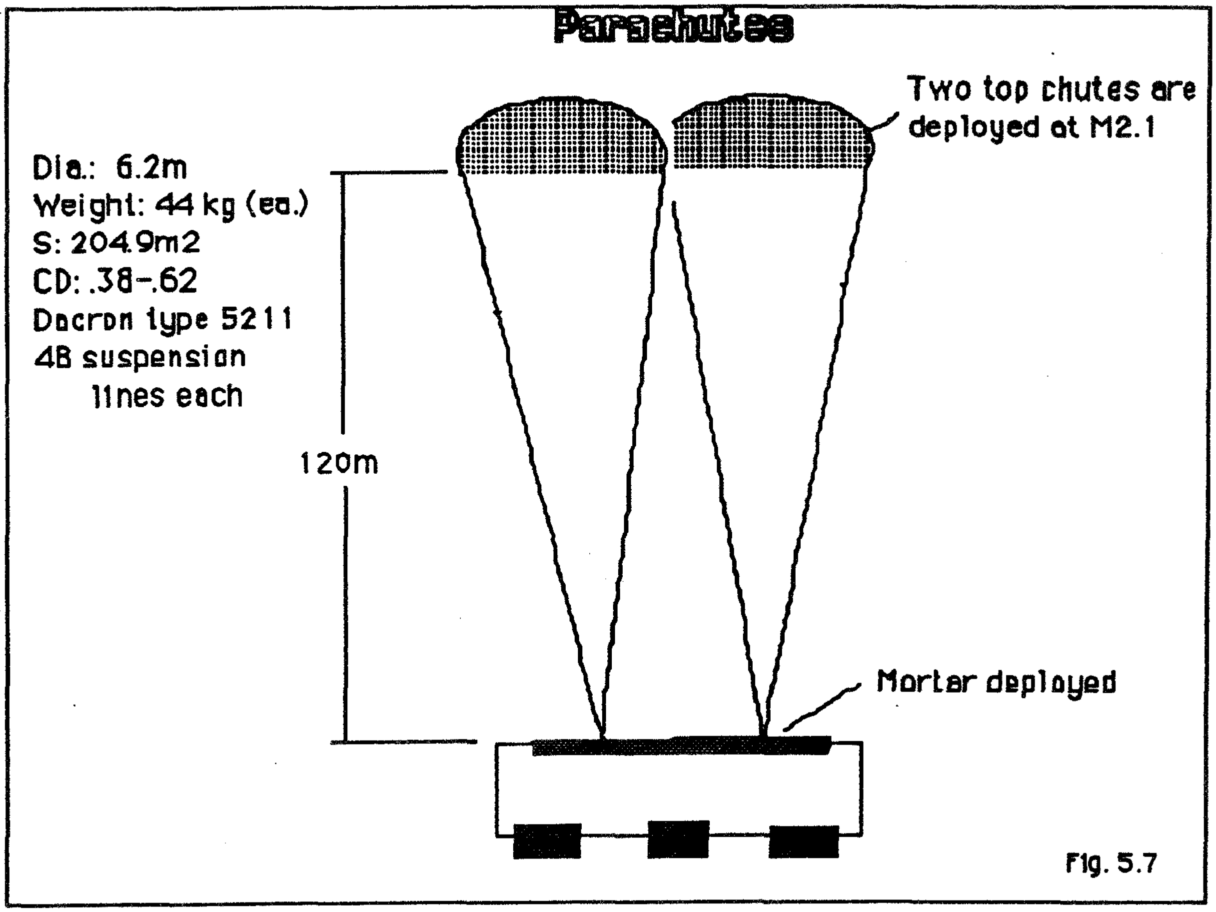

Fig. 5.7

Landing

The landing sequence consists of a final deceleration of the 2100 kg lander to a vertical velocity of less than 2.44 m/sec and a horizontal velocity of negligible value (see PPS for retro information). The landing gear, shown in figure 8, uses the same component design used on the older Viking missions. It utilizes a crushable honeycomb structure in the shock absorbers to dissipate most of the shock upon impact. At the end of each landing strut is a wide footpad to eliminate severe embedding in compressible soils. The placement of the six landing components is designed to distribute the impact stresses equally along the cargo bay and to provide a stable support. Movement of the landing strut will automatically shut down all descent engines[7]. Further protection for the payload against stresses will be provided by hydraulically operated shock-absorbing payload mounts located within the cargo bay.

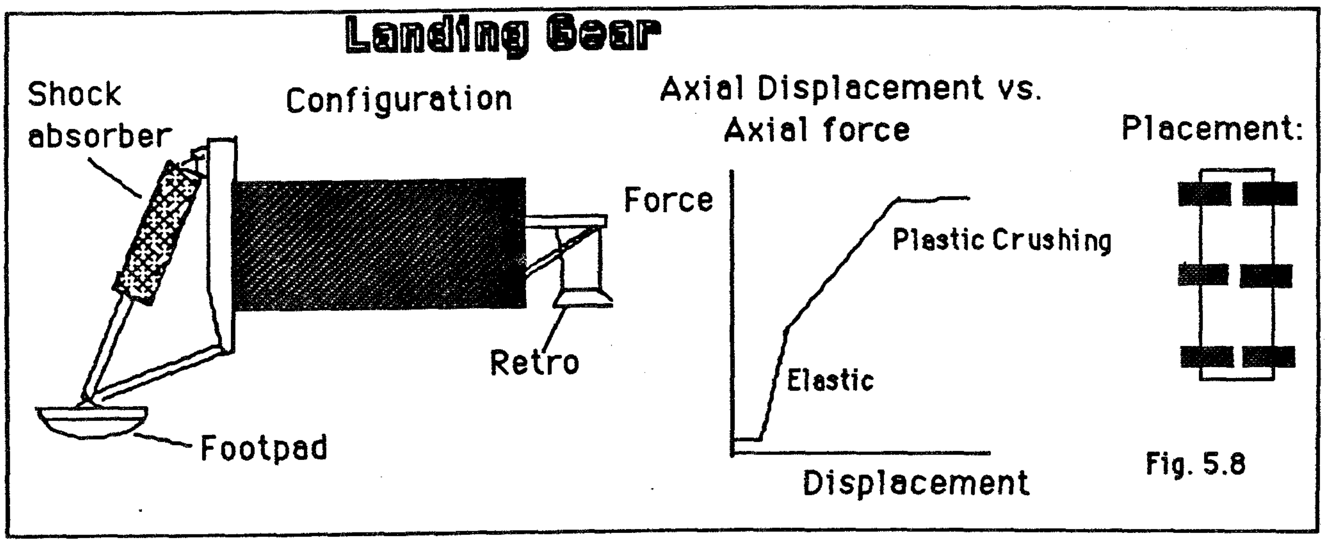

Fig. 5.8

Problem Areas

There are several aspects of the mission that deserve further testing and analyzation. Although the shield size has been drastically reduced from the spherical case, it is still a very large component to deal with in terms of structural support against aerodynamic loading and assembly upon spacestation arrival. Another problem area also associated with the magnitude of our payload is that of excessive parachute loading. The mass of our lander is roughly twice that of the Viking lander and our suspension lines are considerably longer. Although we are using two parachutes, more suspension lines and improved fabric may be needed to cope with higher loads.

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Conclusion

By utilizing aerobraking techniques, employing an elliptical shield, substantial fuel and mass savings of approximately 30% (calculated by T. Horton, PPS) can be made without added cost or complexity. Separating the reentry vehicle and the orbiting satellite at the exact satellite altitude will increase mission efficiency. A parachute and landing gear system utilizing proven Viking technology will safely deliver the payload to the Martian surface at minimal cost

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Information on SRB separation subsystem and final Viking parameterswas provided by Micheal Lembeck.

Satellite Power System

Todd Allyn Horton

The requirements as spelled out in the Request for Proposals are that the power system must last at least four years after reaching Mars, it must be simple, reliable, low cost, previously proven, and available by 1998. Derived requirements are: the system must send telemetry to, and accept commands from CDC, the system should provide for control outputs and sensor inputs, it should be self-powered, and the power system should provide an uninterupted power supply that is protected against power surges to the bus subsystems.

The estimated loads were gathered from the various subsystems. These loads were 100 watts for CDC, 150 watts for AACS, 77 watts for SCI, and 200 watts for PPS. Thirty-three watts is allowed for losses and use by the power system. The total maximum power needed is 560 watts.

Power sources considered were solar cells/batteries, dynamic systems, and RTG's. For this application, RTG's and dynamic systems are much heavier than solar cells/batteries as shown in Figure 6.1. Solar cells and batteries are the only sources that were seriously considered.

Solar cells and battery combinations have been used with success for years. They are reliable, but expensive. Two types of batteries were considered, NiCd and NiH_2. The main parameters and results of these are summarized in Table 6-1. A bus voltage of 35 volts was assumed. The actual voltage will depend on existing power processing units used. The time in Mars' shadow was given as .5315 hours by MMPC. The method for calculations is as follows:

number of cells=$P_L T_E$/DOD

T_E=time in shadow=.5325 hour

P_L=power load=560 watts

DOD=depth of discharge

mass of battery=(# of cells)(mass/cell)

ampere hour capacity, c=$P_L T_E$/{DOD(V)}

V=bus voltage=35 volts

	NiH2	NiCd
Maximum DOD	57%	45%
Energy Density	44.8Whr/kg	30Whr/kg
Energy per Cell	53.6 Whr/cell	32 Whr/cell
Mass per Cell	1.436 kg/cell	1.07 kg/cell
Number of Cells	10	21
Total Mass	14 kg	22.1 kg
Ampere Hour Capacity	14.92 Amp-hrs	18.9 Amp-hrs

Table 6.1 NiH_2-NiCd comparisons

The solar constant at Mars is 0.827 cal/[cm^2 min]. An operating temperature of 40^o C was assumed. Solar cell degradation was assumed to be 30% in 5 years. A packing factor of .88 and efficiencies of 12% to 25% at 25 degrees was used. 12% efficiencies are common today, but 25% efficiency should be possible by 1998. The efficiency drops 0.5% for each degree drop in temperature. An array area density of 3.18 kg/m^3 was assumed. Solar array sizing was accomplished by using the method that follows.

P_T=P_L +cV/N

P_T=total power

c/N=sunlit charging current (amps)

$N=T_S/DOD$

T_S=time in sun=3.72 hours

power required=$P_{BOL}=P_T$/time degradation

time degradation=1-.3=.7

area $A=P_{BOL}$/[(solar constant)(packing factor)(efficiency)

(1-temp degradation factor)(operating temperature -25))]

solar array weight=A(array density)

Solar cell sizing is summarized in Table 6.2 for both NiCd and NiH_2 combinations.

	NiH_2	NiCd
solar array area at 12% efficiency	14.10 m2	14.6 m2
solar array mass at 12% efficiency	22.6 kg	23.4 kg
solar array area at 25% efficiency	6.77 m^2	7.0 m^2
solar array mass at 25% efficiency	10.8 kg	11.2 kg

Table 6.2 Solar cell size and mass with the two battery types

From the above studies, the NiH_2 battery was chosen. The NiH_2/solar array system is 8.5 kg less than the NiCd/array system.

The lander could be in orbit for as long as a week. Since the lander fuel will not be heated by the sunlight, the tanks must be heated. Also, 50 watts for CDC is estimated. The estimated total power load for the lander is 100 watts. To achieve this power load, a small NiH_2/array system was used. Again, T_S (3.72 hrs) and T_E were given by MMPC. The lander requires 3 cells with a total mass of 4.3 kg. The area of the array is 2.03 m^2 and has a mass of 6.5 kg.

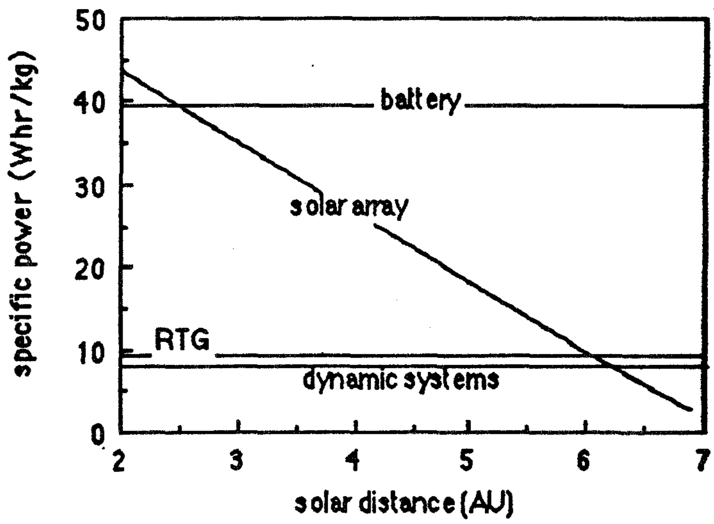

Figure 6.1 Power source specific power

PROPULSION (HIGH THRUST)

Requirements from RFP are the same as for power. Derived requirements are that the system needs to send telemetry to and receive commands from CDC, it must allow for sensor inputs such as temperatures, pressures, electrical loads, and structural loads, and it must also allow for control outputs such as power relays and valve actuations. The propulsion system is required to supply the delta-V needed to get to Mars, accomplish aerobraking, place the satellite in the desired orbit, and land the payload on the surface. In addition, the system must supply small delta-V's for AACS.

Mass estimations were gathered from all the various subsystems through STR. The minimum delta-V required to leave low Earth orbit is 3628 m/s. After this minimum, the delta-V increases slowly. The propulsion system is designed to work for several launch windows with little or no modifications to the design for the higher energy missions.

At Mars, 5 burns are needed in addition to any corrective burns that may be needed. The first places MARTIAN in an elliptical orbit about Mars. This requires a delta-V of 615.1 m/s. A second burn, delta-V of 1.06 m/s, decays the orbit for the aerobraking. During the fifth orbit of aerobraking, the satellite is boosted into its desired orbit, requiring 92.45 m/s. Upon completion of the aerobraking maneuver, the engines fire to circularize the orbit, 172.2 m/s. The final burn begins the landing sequence. This delta-V was calculated to be 51.6 m/s. All delta-V's at Mars except burns 2 and 5 were determined by AERO.

After several runs of AEROB (described by AERO), it was determined that the most efficient way to do these burns is: A liquid chemical engine supplies the 615.1 m/s burn. This engine is also used during the Earth to Mars transfer for corrective burns. After the first burn at Mars, the engine, tanks, and supporting structure are jettisoned. The second burn is performed by the same engine that is used to boost the satellite. A third engine is used for the final 2 burns.

The fuels considered were hydrogen, hydrazine, unsymmetrical dimethylhydrazine (UDMH), and monomethylhydrazine (MMH). Hydrogen gives very high performance but is cryogenic and thus cannot be stored for long periods. It can, however, be used to send MARTIAN on its way. Hydrazine, UDMH, and MMH all have similiar properties. All three react with many materials but do not react with stainless steel or aluminum. Hydrazine is an excellent monopropellant when decomposed by a catalyst. The simple feed system of monopropellant hydrazine can be extremely reliable. The freezing point of pure hydrazine is 274^{o} K. Both UDMH and MMH have a lower melting point (around 217^{o} K) and higher boiling point (335^{o} K) than hydrazine. Burning UDMH and MMH with an oxidizer produces an I_{sp} slightly lower (1-2%) than burning pure hydrazine with the oxidizer. UDMH can be mixed with hydrazine (up to 50%) . All hydrazines are toxic with UDMH being the least toxic and MMH the most. Freezing has no effect on hydrazine, UDMH, or MMH, but freezing a UDMH/hydrazine mixture requires a special remixing process.

Oxidizers considered were nitric acid, nitrogen tetroxide, oxygen, and fluorine. Nitric acid is highly corrosive but can be stored for long periods of time in stainless steel containers. Adding a small amount of fluorine ion makes it less corrosive. Nitric acid freezes at 231^{o} K and boils at 356^{o} K. A high specific gravity (1.55 g/cm^3) allows for compact vehicles. Nitrogen tetroxide is the most common space storable oxidizer. It is easily frozen (264^{o} C) and has a high vapor pressure, requiring it to have heavy tanks. Nitrogen tetroxide does have a fairly high specific gravity of 1.44. Oxygen and Fluorine are cryogenic and therefore unsuitable for long missions. They are, however, excellent oxidizers and could be used with hydrogen to produce I_{sp}'s as high as 500 seconds. Fluorine is extremely hard to handle and is not readily available.

The temperature of space is 3^{o} K. Any surface shadowed from the sun will be at this temperature. The temperature of a black body in the sunlight is given by (solar constant)=(Boltzmann constant)$(T^4-T_o^4)$

Boltzmann's constant = $5.67(10)^{-8}$ W/(m^2K^4)

At Earth, the solar constant is 1.920 call/cm^2/min and gives a skin temperature of 392 K. At Mars it is 317^o K. The propellants must be kept from boiling or freezing at both Earth and Mars. All propellants considered have the possibility of freezing. Since AACS has determined the craft will not be spin stabilized, the tanks will require heating. 200 watts is estimated to run these heaters and any valve actuations needed.

The equation relating delta-V to I_{sp}, propellant mass, and vehicle mass is delta-V = $I_{sp}(g_o)$Ln(initial mass/final mass). Staging reduces the total mass needed to achieve the same delta-V.

The first burn at Mars is the highest and requires the most propellant. Since the higher the I_{sp} the less propellant is used, this first burn should have as high an I_{sp} as possible. Figure 6.2 shows the propellant mass required verses Isp for this burn. The hydrazines and nitrogen tetroxide have an I_{sp} of around 293 seconds. Using nitric acid as an oxidizer raises the I_{sp} to 300 s. Pure hydrazine will not be used in the primary engines since it freezes too easily.

To reduce costs, an engine that has already been designed will be used. Nitrogen tetroxide/50%UDMH-hydrazine combinations have been used with success in the past. These propellants were chosen for the 2 primary engines used after Mars capture. The circularization burn (delts-V=172.2m/s) will be done by an engine that produces 4000 N of thrust. This is the minimum thrust that will accomplish the burn in 130 seconds, which is 2% of the time of the orbit

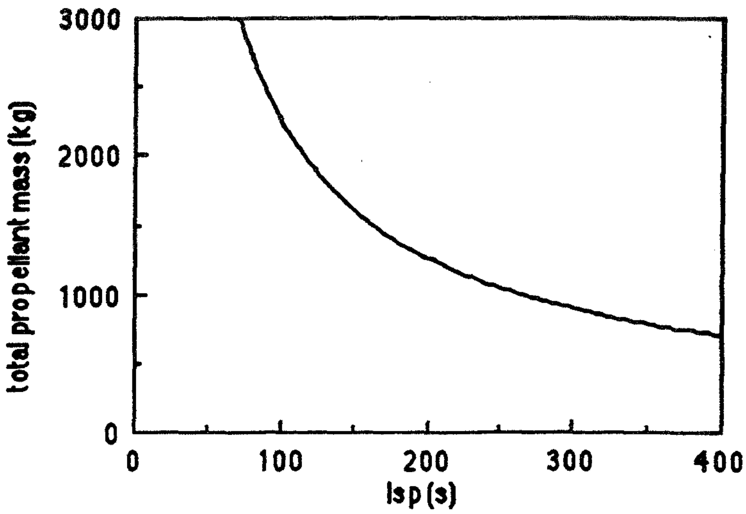

Fig. 6.2 propellant mass vs. Isp

and the maximum time to assume an impulsive burn. The mass of this engine is 60 kg including everything but tanks. The dimensions are shown in Figure 6.3. These dimensions were estimated by finding the mass flow for I_{sp}=293s (I_{sp}=Thrust/mass flow). From this the area of the throat can be found using the equation for maximum mass flow through a nozzle.

$$dm/dt = A_t P_o k\{[2(k+1)]^{(k+1)/(k-1)}\}^{1/2}$$

At = area of the throat
Po = chamber pressure = 500 psia
k = 1.24

The exit velocity is determined from the thrust and mass flow using F=(mass flow)(V_e). Now the exit area can be found by using the isentropic equations to find the exit temperature T and then the exit area A_e.

$$T=To-(k-1)V_e^2/2kR$$

R = gas constant =21.8 g/mol

$$A_e=(A_t/M)\{[2/(k+1)][1 + (k-1)M^2/2]\}^{(k+1)/[2(k-1)]}$$

M = exit mach number=$(kRT)^{1/2}$

The nozzle is conical with apex angle of 15^o. The thrust chamber is about half as long as the nozzle and the radius is 5 times the radius of the throat. All calculations assume a chamber pressure of 500 psia and 2600^oK chamber temperature. The oxidizer to fuel mixture ratio is 1.81 by mass and 1.125 by volume. The average specific gravity is 1.195 g/cc. The ratio of specific heats of the gaseous products is 1.24. AERO gave the propellant mass for circularization to be 181.16 kg. The propellant mass needed to enter the atmosphere is 55.74 kg. The total mass is 260 kg including 10% extra. Of this, 92.7 kg is fuel and 167.3 kg is oxidizer. The tank volumes are $.1021m^3$ for fuel and $.1155m^3$ for oxidizer.

The engine that starts the aerobraking and boosts the satellite will produce 1500 N. The mass will be 40 kg including everything but the tanks. The dimensions are given in Figure 6.4 and were determined the same way that the 4000 N engine was. The first burn, 1.06 m/s, requires a propellant mass of 1.37476 kg and lasts 2.6 seconds. 13.08 kg are required to boost the satellite (92.45m/s). The mass of the satellite including fuel and booster is 400 kg. Using the same chamber conditions and mixture ratio, the total mass of fuel is 5.66 kg and is stored in a 0.006 m^3 tank. The mass of oxidizer is 10.24 kg and requires 0.007 m^3. The burn lasts 25 seconds.

The engine used to enter Mars orbit will be an Agena 8096. This engine has a reliability of 99.8% and used hypergolic UDMH and nitric acid. The engine is rated to burn for 240 seconds and produces a thrust of 71170 N at an I_{sp} of 300s. The Agena is described by Figure 6.5.

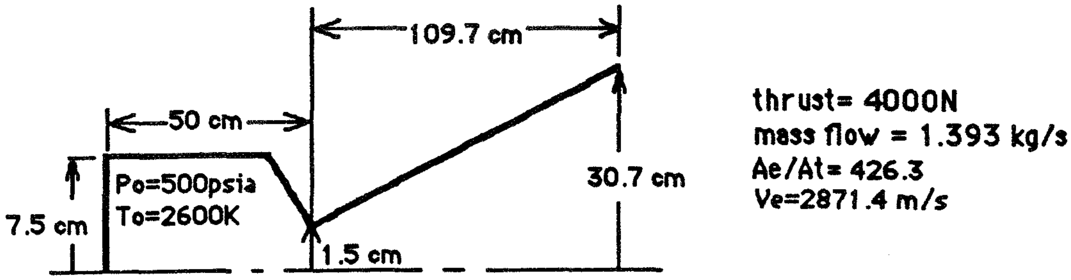

Fig. 6.3 4000 N engine

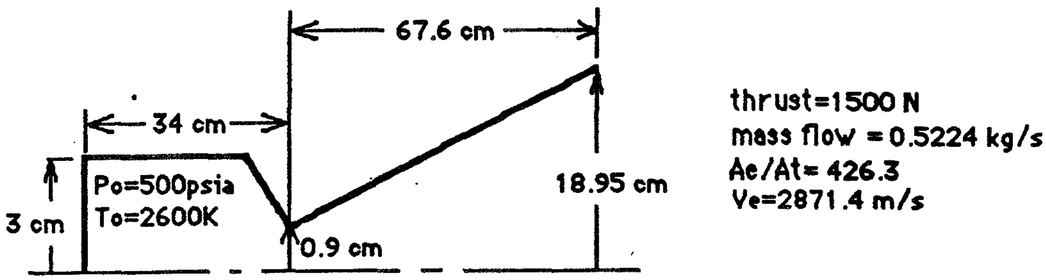

Fig. 6.4 1500 N engine

The Mars capture burn requires the Agena to burn for 37.4 seconds. This leaves 3.47 minutes to be used during the trip and at Earth departure. If 2 identical Agena stages are used, a maximum delta-V at Earth of 4200 m/s can be achieved. This does not include burning the propellant needed for capture. Using the 2 identical Agena stages will be less expensive than custom designing a booster and orbit insertion engine. In addition, the stages provide a maximum delta-V of 4200 m/s allowing a flexible launch window.

The mixture ratio was found by stoichiometric combustion of nitric acid and UDMH. This mass ratio is 3.8. The mass flow through the Agena is 24.2 kg/s. For a maximum of 240 seconds, the propellant mass is 5808 Kg. The mass of

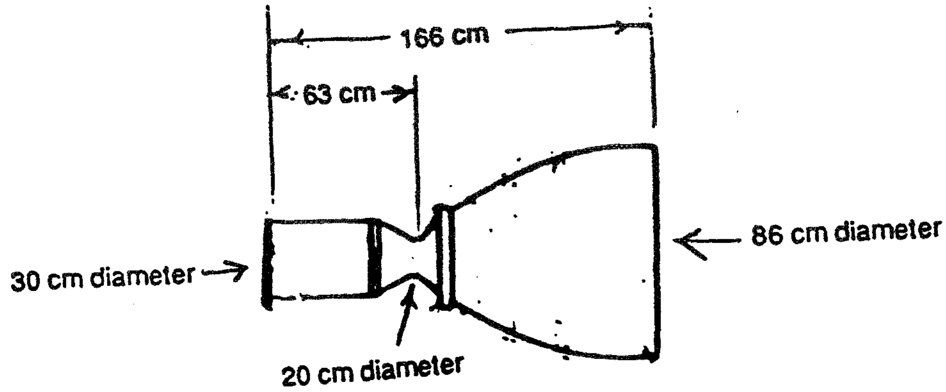

Figure 6.5 Agena 8096

the dry engine is 133 kg. The maximum UDMH and nitric acid the engine can burn is 11.62 kg and 4646.4 kg respectively. These masses lead to volumes needed for the nitric acid and UDMH (3 m^3 and 1.66 m^3, respectively).

The propellant mass needed to enter Mar's orbit (from AERO) is 905.6 kg. It was assumed that a delta-V of 300 m/s would be used for corrections on the way. This leads to an additional propellant mass of 515.66 kg, for a total of 1421.3 kg. 1027 kg of this is UDMH and 4107 kg is nitric acid.

With these masses, the first stage Agena produces a delta-V of 1979.3 m/s. The second stage produces the remainder of the delta-V needed and uses 4272 kg of propellants, 1020 kg less than the maximum that the Agena can burn.

. AERO requested the mass savings of aerobraking over the all propulsive case. The delta-V required to enter the desired orbit at Mars is 1994 m/s was given by MMPC. The mass at the start of aerobraking is 4800 kg and results in 3500 kg in orbit including the reentry heat shield. Using a single stage with an I_{sp} of 300 s and neglecting the mass of the booster used at Earth, the mass in Earth orbit is 16488kg. 11688 kg of fuel is needed to leave Earth. For the all propulsive case, the total usable mass in Mars orbit is 3500 kg including heat shield. The mass of propellant needed to enter orbit is 3396 kg. This gives the total mass to be boosted from Earth of 6896 kg. The propellant needed to boost this (Isp=300s, massless booster) is 16792 kg. The mass in Earth orbit is 23688 kg.

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Applicable RFP Requirements: see PPS section 6

Lander's Reentry System

The first step in trying to size the lander's retro-rockets is to determine the craft's terminal velocity due to the drag induced by its parachutes. Terminal velocity will be described by the equation (AEROB):

$$\text{drag force} = 1/2 * p * V^2 * A * C_D = M * g_m$$

where: p = density of Martian atmosphere

V = terminal velocity

A = area of parachute

C_D = coefficient of drag

M = Lander entry mass

g_m = gravity of Mars = 3.725 m/s^2

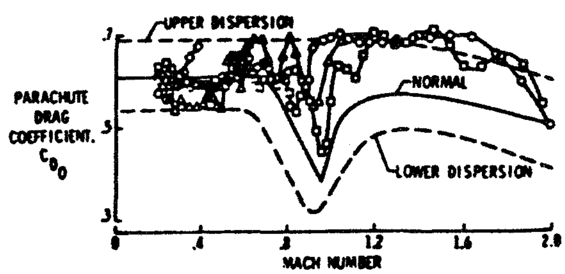
Unfortunately the coefficient of drag and the density of atmosphere change throughout the lander's descent. Thus for the ease of approximation I will assume:

$p = .012$ kg/m^3 and $C_D = .6$

Figures 7.1 and 7.2 illustrate that these guesses are reasonable for a greater majority of the entry sequence.

Altitude (km)	Temperature (°K)	Density (kg/m^3)	Scale Height (km)
0	214	1.56 E-2	
10	205	6.47 E-3	11.36
20	188	2.63 E-3	11.11
30	175	9.81 E-4	10.14
40	162	3.40 E-4	9.44
50	152	1.08 E-4	8.72
60	144	3.19 E-5	8.20
70	140	8.75 E-6	7.73
80	139	2.29 E-6	7.46
90	139	6.03 E-7	7.49
100	139	1.60 E-7	7.54
120	136	1.60 E-8	8.69
130	120	3.80 E-9	6.96
140	166	7.25 E-10	6.04
150	182	2.41 E-10	9.08
160	186	9.35 E-11	10.56
170	167	4.21 E-11	12.53
180	148	1.80 E-11	11.77
190	143	6.62 E-12	10.00
200	102	3.30 E-12	14.36

Fig 7.1 Mars Atmosphere



Parachute drag coefficient compared with steady-state envelope.

Fig 7.2

Using the work done on the Viking Lander's parachutes, some good estimates can be made about parachute size and weight:

Viking Parachute area = 204 m^2 and weight = 44 kg

From this data we can estimate that the parachutes will have a specific weight = 0.2147 kg/m^2. This area also gives us a place to start guessing from to determine the terminal velocity. Now we must look at the Rocket Equation (which includes gravity) to determine the mass of propellant needed by the landing rockets:

$\Delta V = Isp * g_o * \ln (Mo/Mf) - g_m * Tb$

where: $Tb = Mp/m$

ΔV = terminal velocity

Isp = 300 sec (high performance hydrazine motor)

m = mass flow rate of propellant

Assume that the lander reaches terminal velocity by 1 km altitude and that at this altitude the lander starts its decceleration burn. Also assume that the lander's terminal velocity will be about 120 m/s. These assumptions along with the following equation will yield a mass flow rate:

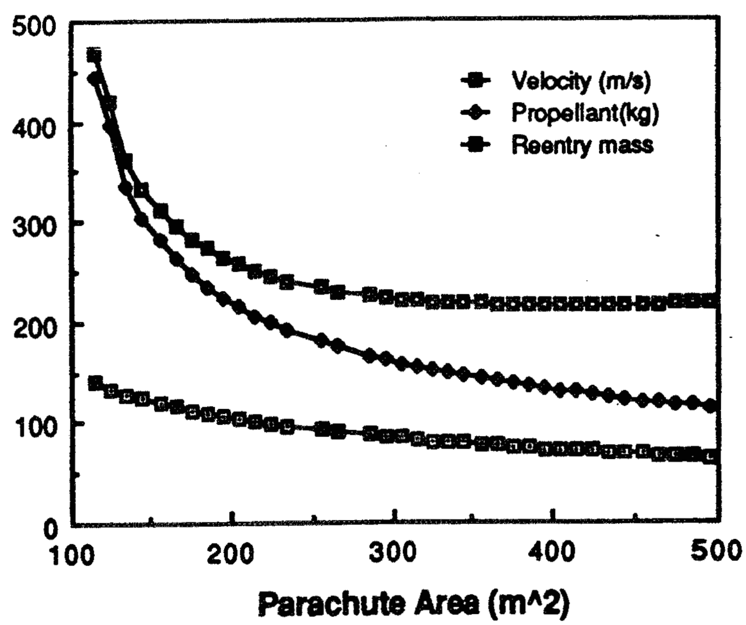
$$Vf^2 = Vi^2 + 2 * A * S$$

$$M * A = M * g_m - T \quad \text{where } T = \text{thrust}$$

$$T = m * Isp * g_o$$

This yields a propellant mass flow rate = 2.36 kg/sec which is reasonable compared with the Viking Lander m = 1.225 kg/sec. Now a computer is used to iterate the Rocket equation to solve for the propellant mass. In addition to this, the area of the parachute is altered to do a trade study between what is the optimum size for the parachutes that will minimize the mass of propellant plus the mass of parachutes. These results are given in figure 7-3

Fig 7.3 **Parachute area v.s. Reentry system mass**



The optimum results are:	The chosen design
parachute area = 384.96 m^2	404.96 m^2
terminal velocity = 74.29 m/s	72.38 m/s
mass of propellant = 135 kg	131 kg
Total mass of system = 215.49 kg	215.79 kg

The reason slightly higher values were chosen is because the parachute area corresponds with the area of two off-the-shelf Viking parachutes. This will decrease cost while only sacrificing .3 kg of weight.

Review of Possible Propulsion System Options and Analysis

Laser Propulsion

This system involves using an external laser to vaporize the propellant of a rocket and then allowing the vaporized gases to accelerate through a nozzle to produce the desired thrust. The amount of energy that can be conveyed in this manner can be large compared to the amount produced during a chemical reaction.

Continuous Wave Laser Propulsion is best suited for missions requiring high levels of specific impulse (1000s) and moderate thrust levels (10,000 N = 2248lb). Most research has ben done on the following two scenarios: Earth launch, and orbit maneuvering missions. Interplanetary missions using all CW laser propulsion don't seem economical because the system would require a space-based laser satellite capable of producing a 1 GW laser beam both at launch and arrival points. I propose a laser propulsion burn at Earth and then revert to the traditional chemical rocket technology for the remainder of mission duties. This seems to be advantageous considering the mission requirement of an aerobraking subsystem which requires a high thrust propulsion system for its maneuvers.

Assumptions in analysis

There exists a satellite in Earth orbit that is capable of producing a 1GW laser beam.

Assuming worst case pointing efficiencies the size of the spacecraft's focusing mirror would have to be 10m in diameter

Isp=1000s and 40% thruster efficiency

Because of the increased complexity of the Laser propulsion stage, assume a structural efficiency = .25

A typical Centaur type rocket stage is used for comparison

E=.1, Isp=444 s

Numerical value for minimum ΔV = 3.6 km/s (from MMPC)

<u>Equations used for sizing analysis</u>

$\Delta V = Isp * G_0 * \ln(Mo/Mf)$

$Mf = Mo - Mp$

$E = Ms/(Ms+Mp)$

$Mo = Ms + Mp + Ml$

Isp= specific impulse

G_0= gravity at Earth= 9.8 m/s^2

Mo= initial mass

Mf= final mass

Mp= propellant mass

Ms= structural mass

Ml= payload mass

E= structural efficiency

Results

Solving the above equations

	Centaur Type Booster	Laser Stage
Mp	12016.92 kg	4167.89 kg
Ms	1201.7 kg	1041.97 kg
Mo	21218.6 kg	13209.9 kg
Mf/Mo	.43	.685

Low Thrust Systems

There are several fundamental problems in calculating low thrust trajectories. Most importantly, the well understood ballistic trajectory analysis does not work because the engines are thrusting most of the time which means that the loss due to gravity will be appreciable. There are no software packages available to do the necessary trajectory calculations, which means there are going to have to be some large assumptions made.

Assumptions:

Martian is launched from a Nuclear Start Orbit (800 km)

Trajectory obtained from reference 1 will be the baseline for the comparative study.

Isp = 4138 s

thrust to weight ratio = T/W = 4.39 E-5

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$\Delta V = 12.919$

thrust time = 298 days

mission time = 467 days

payload mass = 5665 kg (our payload mass is 4800 kg)

assume that all ballistic equations work within these parameters

Nuclear Electric Rocket Propulsion (NEP)

The first step in the analysis of a nuclear powered system is to determine what is the size and power output of reactors in existence or proposed. These systems are often referred to as Space Nuclear Power Systems (SNPS).

The SP-100 project was established to develop and demonstrate the feasibility of space nuclear power systems that produce power in the 100kW - 1MW range. This project chose the 300 kW power level for extensive development. This power source is a fast-spectrum reactor fueled with Uranium Nitrate and cooled with liquid Lithium. These are the specifics on the design:

Fig 7.4 Conceptual Design Nuclear Power System Characteristics

Total Delivered Electrical Power	300 kW_e
Delivered Voltage Range	100-800 V_{dc}
Total System Mass	7500 kg
Full Power Lifetime	> 7 yrs
Total Useful Lifetime	> 10 yrs
Radiator Heat Removal Capacity	6.3 MW_t
Conversion/Radiator System Efficiency	6.8%
Thermoelectric Converter Material	LaS_x
Total Number of Converters	6×10^5
Heat Pipe Working Fluid	Lithium
Total Number of Heat Pipes	320
Radiation Shield Cone Half Angle	$15^\circ - 20^\circ$
Reactor Thermal Power	7.0 MW_t
Reactor Fuel	UN

NEP systems can be broken up into three broad categories: Electrothermal thrusters, Electrostatic thrusters, and Electomagnetic thrusters.

Electrothermal thrusters encompasses all methods whereby a propellant gas is heated electrically and then expanded through a nozzle to convert the thermal energy into kinetic energy.

Resistojet thrusters heat the propellant flow by passing it over an electrically heated solid surface. In this way Earth-storable propellants can deliver similar performance to high energy bipropellants. Performance of this system is greatly affected by the use of cryogenic fuels or an increase in the inputed power. This system is well developed in the low power range and is currently being used in station-keeping applications.

Arcjet thrusters heat the propellant by passing it though an arc discharge. Their operation is best illustrated by fig 7.5. Arcjet thrusters are the most mechanically simple of all types of thrusters.

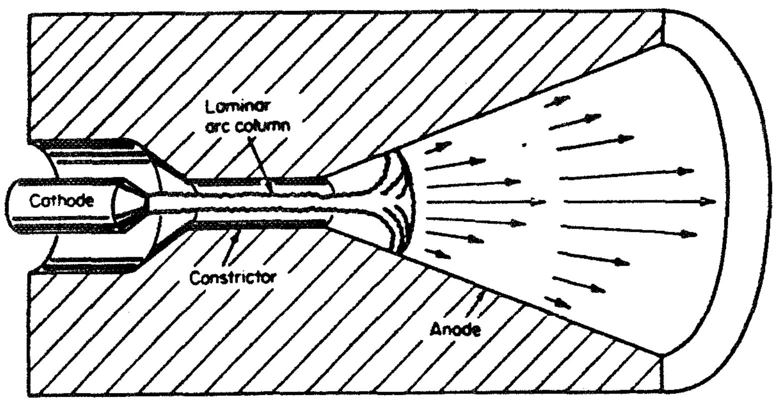

Fig 7.5 Arcjet Thruster

Electrostatic Propulsion , also called Ion thrusters, extract propellant ions from the discharge chamber plasma and accelerate them electrostatically to form a high velocity ion beam. Electrons are then injected into the ion beam to neutralize the charge of the beam so that the spacecraft maintains a neutral charge.

Electromagnetic , also called magnetoplasmadynamic (MPD), thrusters work by accelerating a body of ionized gas through the interaction of the electrical currents being driven through the gas, with magnetic fields established either by the electrical currents or by external means. The most developed thruster of this type is the pulsed MPD.

Analysis:

For reference purposes let us review the state of technology now and in the future before 1998.

Resistojet	Input Power (kW)	12.3	30	100 *
	Propellant	NH_3	H_2	N_2H_4
	Thrust (N)	2.93	6.04	140
	Isp (sec)	423	846	1030
	Thrust Efficiency	.5	.85	.7
Arcjet	Input Power (kW)	30	30	30 *
	Propellant	H_2	NH_3	NH_3
	Thrust (N)	3.35	2.35	2.8
	Isp (sec)	1010	967	1050
	Thrust Efficiency	.54	.38	.45
Ion Drive	Input Power (kW)	6.38	42.85 *	32 *

	Propellant	Xe	Xe	Hg
	Thrust (N)	.24	1.018	.9
	Isp (sec)	3760	4138	5978
	Thrust Efficiency	.71	.81	.79
MPD	Input Power (kW)	195	87.5	1781 *
	Propellant	H_2	NH_3	NH_3
	Thrust (N)	2.22	.95	3.3
	Isp (sec)	4520	6500	5200
	Thrust Efficiency	.254	.344	.40

* = indicates predicted performance values

Without going into an in depth analysis of each type of propulsion system, NEP systems can be ruled out by looking back at the mass of the reactor. The mass of just the reactor is 7200 kg plus an extremely conservative estimate of the propulsion system mass equal to 7000 kg leads to a total system mass of 14,200 kg. This is about the same as the weight for the chemical stage. In this case the chemical propulsion system is the clear choice. It has shorter trip times, more reliability, proven technology, uses off-the-shelf hardware, and is still fairly simple. Note: Typically the electrical propulsion systems are better for missions that don't have large time restraints. This missions requirement of having an aerobrake system greatly decreases the benefits of low thrust systems because of the necessity of a chemical stage once reaching Mars.

Solar Electric Propulsion (SEP)

Solar electric propulsion works using ion engines powered by electricity produced by solar arrays. The thruster of choice in this system is the 5 kW Xe ion engine. Using the assumed data from the low thrust trajectory, the rocket equation, along with the following sizing equations we can determine the mass of the SEPs system.

$$82.16 + .3354 * Mp + 19.67 * P = \text{Dry Mass}$$

where Mp = 1000 kg

This gives the total system mass = 2840 kg. Sizing the solar cells from PPS -6 you get an area of 169 m^2. This represents a considerable decrease from the chemical stage, however weight isn't the only factor. The chemical system has decided advantages in some of the key RFP requirements. It is proven technology, uses off-the-shelf hardware, has no real pointing requirements during flight, simpler structure, and attitude and control, simple, reliable, and shorter flight times.

Conclusion

The RFP requirement of having to do an aerobraking maneuver effectively kills the huge weight advantage that some of the electrical propulsion systems would have ordinarily. The one thing that did seem to be decidedly better was the laser propulsion first stage. The only thing that stopped our group from going with it was the uncertainty of a 1GW laser satellite in Earth's orbit.

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STRUCTURES

The Structure subsystem (STR) is responsible for tying the whole spacecraft together, its components being materials and fabrication/assembly. The mission requirements that apply specifically to the STR subsystem distill into the following requirements. First, Structures must provide support for all of the spacecraft components. This includes the need for appendages, such as booms and solar arrays, and the deployment thereof. Secondly, the STR subsystem is responsible for the layout of the spacecraft, minimizing conflicting requirements. This involves the layout of the components of the satellite (science instruments, various sensors, thrusters, tanks, etc.) and the layout of the spacecraft as a whole, from launch from Earth to landing the plane on the Martian surface. Thermal control and launch vehicle compatibility are also issues for Structures.

Of course, STR must also meet the general mission requirements that apply to all subsystems. In all, simplicity, reliability, minimum mass and low cost must be stressed. The components should use off-the-shelf technology wherever possible and future technology which is expected to be available no later than 1998. More specifically, the spacecraft must have a "handle" for retrieval by the remote manipulation arm of the STS space shuttle or similar device on a space station; the extent of space station support (e.g. on-orbit assembly) must be minimized. And finally, the design of the spacecraft should not preclude it from carrying out vastly different missions, while having a design lifetime of at least four (4) years.

Interaction with other Subsystems and Method of Attack

The STR subsystem interacts with all other subsystems as shown in figure 8.1, simply because STR provides support to all spacecraft components. The first consideration is, of course, the mission and subsystem requirements. Then, STR needs to know what is going to be on the spacecraft; all component masses, dimensions, moments of inertia, thermal restrictions, etc. are needed in order to layout the spacecraft and design a workable structure. These component data come from each of the other subsystems and from the Aircraft groups. Orbit determination from MMPC is needed in order to conclude whether or no solar radiation will be a factor. From CDC, SCI and AACS, pointing requirements for the various instruments are needed. Once the data is compiled, AERO and PPS, in addition to STR, need to know the mass breakdown of the spacecraft, while the AACS subsystem needs to know moments of inertia of various spacecraft configurations.

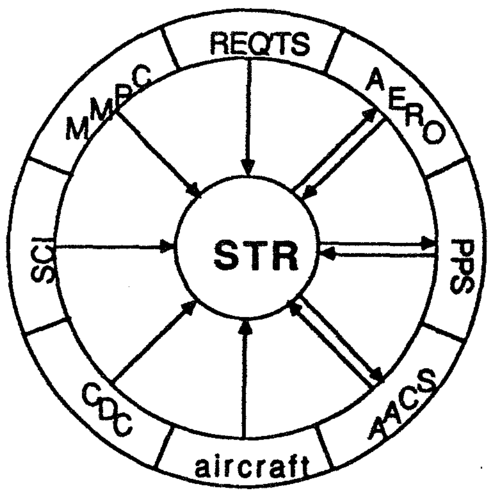


Figure 8.1: Interaction of STR with other subsystems

Once a qualitative layout of components is done, a quantitative layout is required. A program such as INERT by Michael Lembeck is useful in this task. Then a structure can be designed to support the components in the layout determined. Materials must be selected, subject to various constraints which are studied in a trade study, for each and every component of the structure. Once the spacecraft structure has been designed, a structural analysis should be done, including static and dynamic spacecraft response. A study of the thermal control also dictates limitations on the structure. If a proposed structure fails under any of the analyses, the spacecraft must be redesigned and re-tested, until the structure performs satisfactorily.

Layout of Components

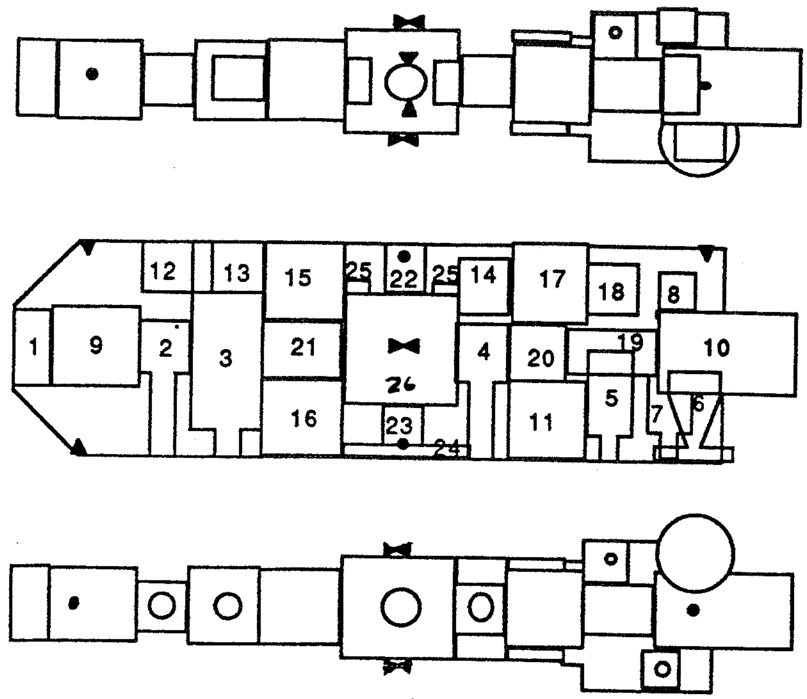
The layout of the components of the spacecraft was done on a modified version of INERT. These modifications were made to INERT to make the layout task simpler. Specifically, the modifications allowed better user interaction and the ability to selectively edit and re-edit the data for a component to view the effective changes. In addition, the modifications facilitated grouping various components into subsystems, so that once a subsystem had been finalized, future changes to other subsystems were simpler and faster.

The first part of the spacecraft to be laid out was the satellite. The gravity gradient boom stabilization (a long boom towards the planet; for MARTIAN, this is balanced by a like boom radially outward from the planet) necessitates that the moment of inertia is maximum about the pitch axis. But, on the other hand, large moments of inertia neutralizes the effects of thrusters and the SEPS solar arrays, when unfurled, increases the other moments of inertia, not effecting that about the pitch axis. So an optimization is desired. The gravity boom frees up the need for a scan platform, so that the science instruments can just be mounted on the vehicle. Specifically, the star trackers and sun sensors of the AACS subsystem has to be able to see out to the stars, while the dust detector looks forward, and the rest of the science instruments, except the magnetometers (which go each on the end of one of the two booms) and part of the radar system, point towards the planet.

The satellite has several dozen components, which makes a quantitative layout quite complex. Figure 8.3 shows the components of the satellite, tabulates their locations and mass breakdown. The resulting satellite inertia matrix is in figure 8.2. Two major configurations of the satellite have been analyzed -- the deployed case (in orbit about Mars) and the stowed case (launch and interplanetary travel). Obviously, the more important of the two is the deployed case, since the satellite must remain stable in Martian orbit for a period of several years.

reentry vehicle inertia matrix:		
73807.8	0	0
0	12677.1	33.99
0	33.99	74145.0
deployed satellite:		
958.24	.0195	-.103
.0195	850.79	.00514
-.103	.00514	180.97
furled satellite:		
160.3	.0195	-.103
.0195	20.85	.00513
-.103	.00513	149.07

Figure 8.2: Inertia Matrices for several Components



1	dust detector	5	0	1.375	0
	planet observers	55.1	-3.6e-4	-.107	-.0778
2	UV photometer				
3	IR Spectrometer				
4	gamma-ray spec				
5	UV spectrometer				
6	radar mapper				
7	atmos. sounder				
8	radio science	8	0	-.8	.225
9	radio	24	0	1.15	0
10	antenna (furled)	5	0	-1.2	0
11	computer	10	0	-.55	-.275
12	accelerometer	1	0	.6	.3
13	accelerometer	1	0	.9	.3
14	accelerometer	1	0	-.3	.225
15	gyro	10	0	.35	.275
16	gyro	10	0	.35	-.275
17	gyro	10	0	-.55	-.275
18	star tracker	8	0	-.8	.225
19	antenna actuat.	6.5	0	-.775	0
20	mapper/ppu	10	0	-.5	0
21	batteries	14	0	.35	0
22	top boom	3	0	0	.3
23	bottom boom	3	0	0	-.275
24	structure	18.7	0	0	-.375
25	sun sensor	2@2	0	0	.25
26	thruster fuel tank	66	0	0	0
	thrusters	28.8	0	0	0
	TOTAL	358.1	-5.6e-5	9.8e-5	9.1e-5

Figure 8.3: satellite layout and partial mass breakdown

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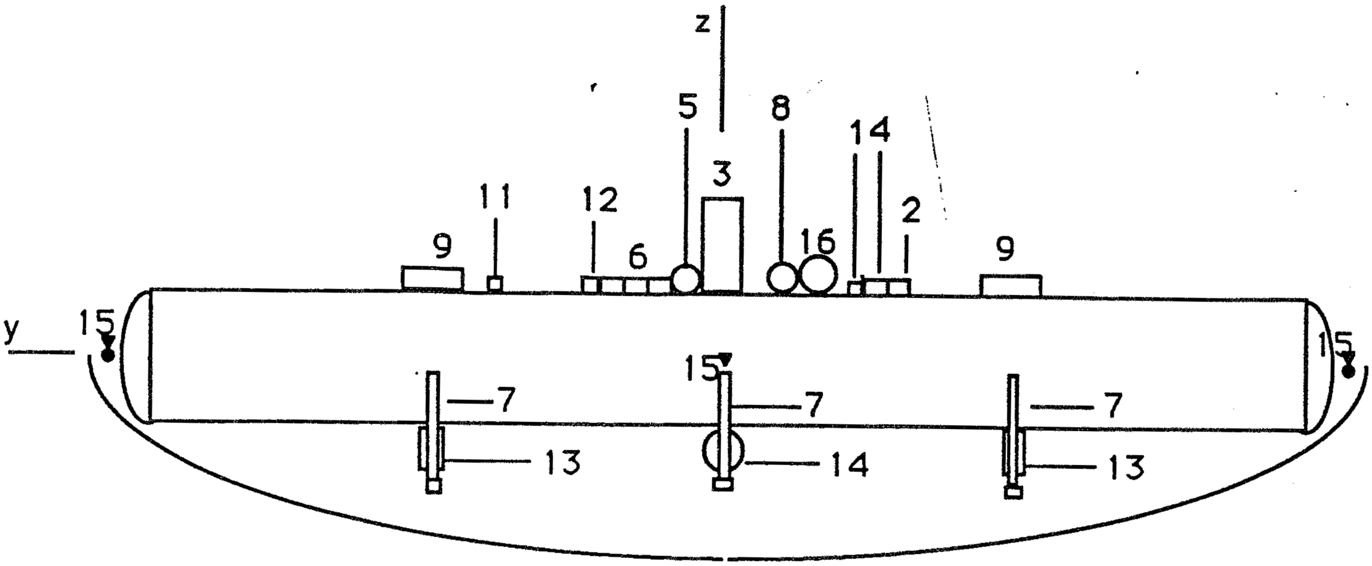

#	Unit	Mass(kg)	Center of mass		
			x	y	z
1	Accelerometer	3@1	0	-2.3	.9
2	Battery	4.3	0	-2.7	.85
3	Booster	60	0	0	1.6
4	Computer	5	0	-2.4	.89
5	Fuel tank	291.2	0	.5	.95
6	Gyro	3@10	0	1.4	.98
7	Landing gear	3@30	0	0	-1.53
8	Oxidizer tank	103.8	0	-.9	.95
9	Parachute	2@30	0	0	.925
10	Payload	1090	0	0	-.3
11	Power processing unit	5	0	4	.9
12	Radio	5	0	1.98	.89
13	Retro	2@25	0	0	-1.6
14	Retro fuel tank	50	0	0	-1.6
15	Thruster	8@2.4	0	0	0
16	Thruster fuel tank	27.5	0	-1.4	.98
	Total	3044	0	0	-.70

Figure 8.4: reentry vehicle mass breakdown

The layout of the reentry vehicle has also been completed. Figure 8.4 shows the aeroshield, payload, and assorted components of the reentry vehicle, in addition to tabulating masses and locations. The resulting moments of inertia are in figure 8.2. The prime requirements for the layout of the reentry vehicle were, again, minumum moments of inertia, and keeping within the wake during aerobraking (which results in a cone of safety; with a circular shield, this cone is two diameters in length; with ~~an elliptical shield, this sh~~ould be two times the smallest frontal dimension, which in the case of MARTIAN, allows a cone of safety 14m long.

The aircraft designs, shown in figures 8.5 and 8.6, are nearly the same in some respects, such as the final total mass, but yet the designs differ widely, for example the issue of using large winglets. While this variance necessitates minor changes in the delivery system, it also adds flexibility to the overall design, facilitating vastly different missions.

Cutting the aircraft into pieces is necessary in order to fit it in the delivery system. Figure 8.7 shows schematically how one of the aircraft designs (Schirle) is arranged in order to maximize the balance of the payload and allow it to be put into a shell for protection against heat from the sun and during aerobraking and reentry.

The vehicle as assembled in low-earth orbit is the combination of the reentry vehicle, the satellite, the satellite booster, and two Agena boosters. The location of the "handle" or "handles" for remote manipulation will be on the structure on the outskirts of the vehicle. Exact location will be chosen according to the ease of access.

Scale → 1:200

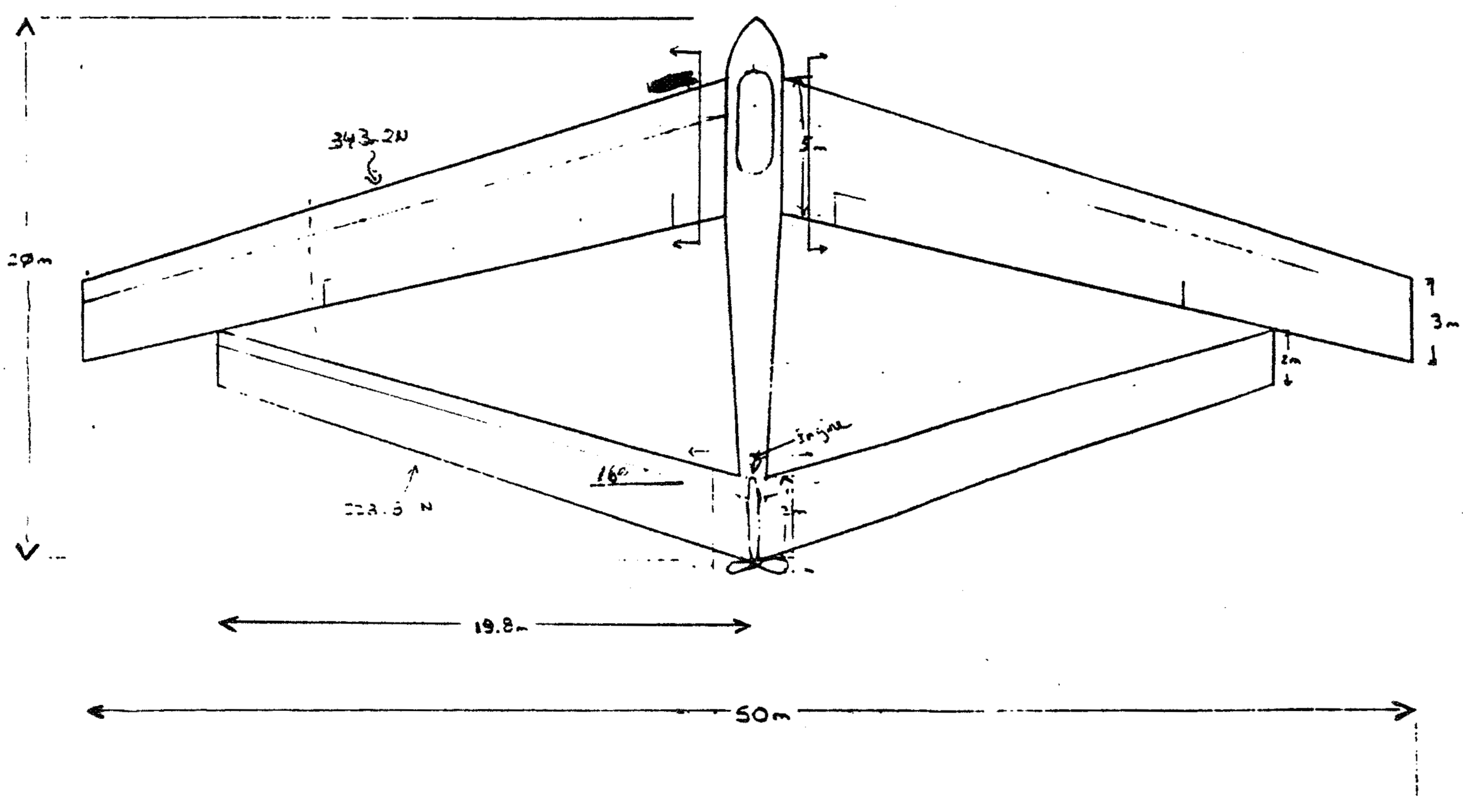

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Figure 8.5: aircraft design (Schirle)

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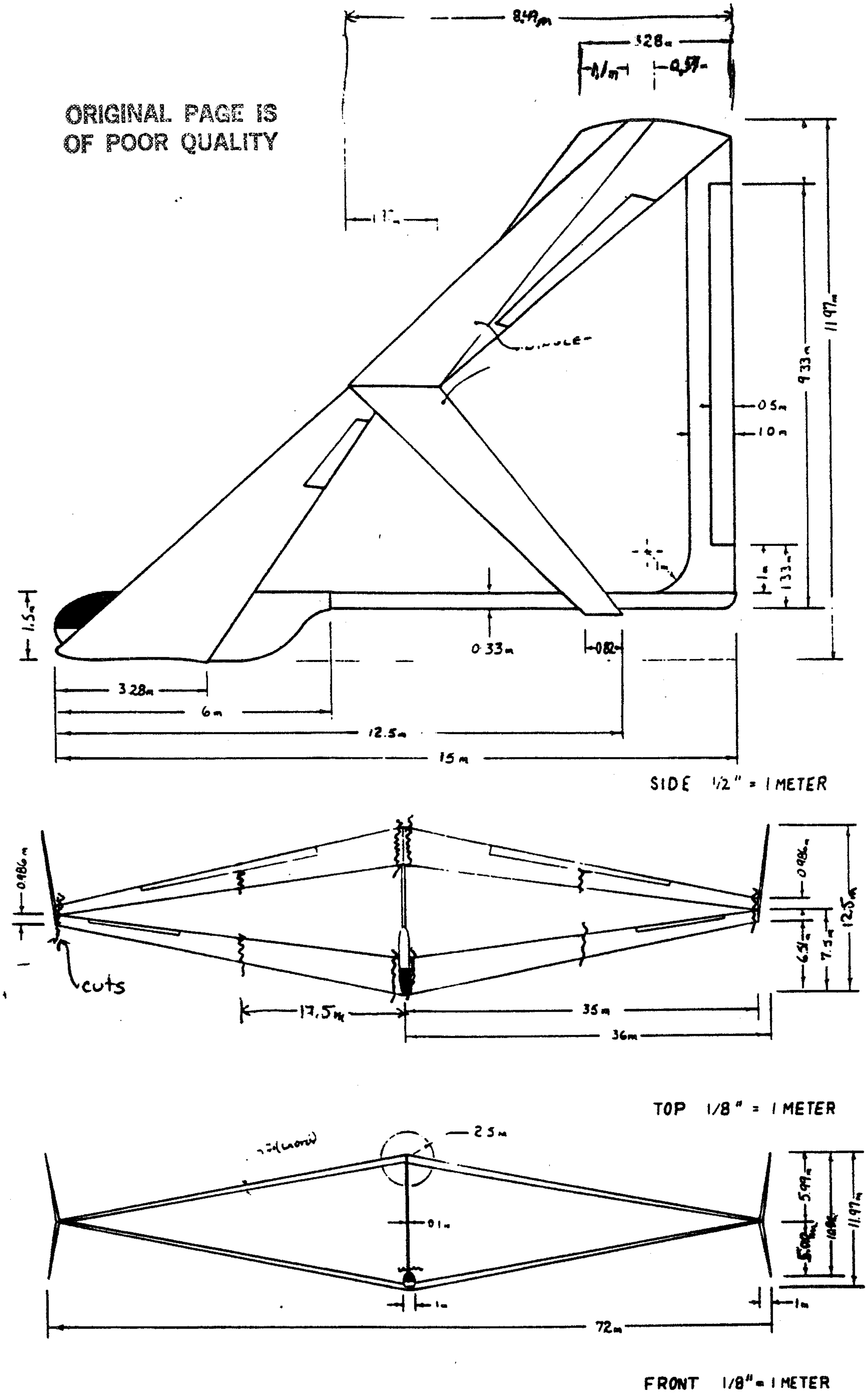

Figure 8.6: aircraft design (Woodruff)

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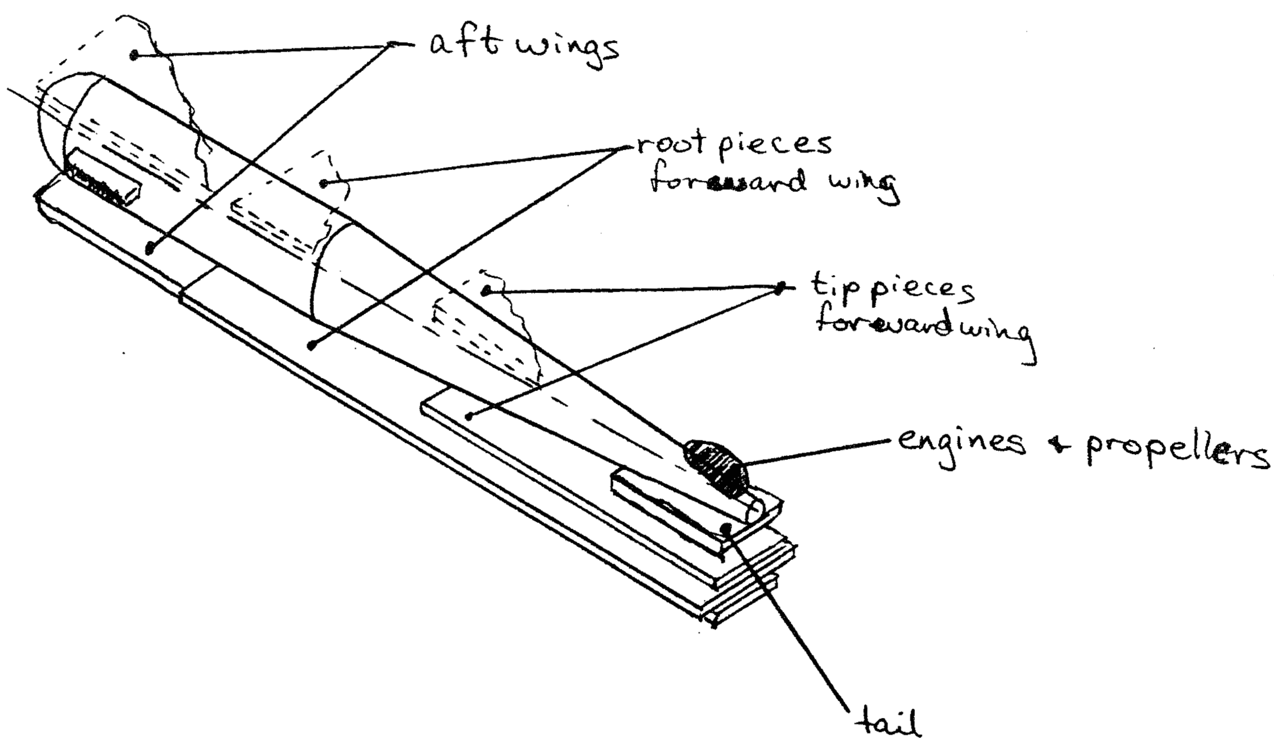

Figure 8.7: arrangement of aircraft pieces

Launch Vehicle Compatibility

The vehicle cannot be launched in any of today's launch vehicles and have no on-orbit assembly. But, a Heavy Lift Vehicle (HLV), such as the "Shuttle C" ("C" for cargo) shuttle-derived vehicle of NASA or the Advanced Launch Vehicle (ALV) of the USAF, would be able to launch MARTIAN with minimal assembly. In fact, with the current requirements for the HLV system, MARTIAN will be able to be completely launched by one HLV, with the whole reentry system intact. Thus the assembly would be limited to bolting on the satellite and the other boosters, which would already be in one piece.

With the shuttle, this simplicity is nowhere to be seen. The aerobrake shield required will not fit in the shuttle in one piece; in fact it will have to be cut into at least four pieces. The aircraft designs also do not accommodate the smaller cargo bay of the shuttle. All this leads to a need for three to five shuttle launches to put MARTIAN into low-earth orbit (LEO) and a need for an incredibly extensive on-orbit assembly. Another factor in the decision on launch vehicle is cost. Currently, STS launches payloads to LEO for approximately $4500 per pound. The required cost for the HLV is "one-tenth that of current means".

The HLV advantages are countered by the fact that the HLV is still in its development stages. Thus any data available today about what the HLV will be like may change. But it is expected to be ready by 1998, and thus will be used to launch MARTIAN.

Materials

Since the requirements for the various parts of the spacecraft are different, multiple material types may be advantageous, but also more costly (there may be bonding problems, for example, between a composite material and a metal alloy). Figure 8.8 shows some of the materials available and some of their advantages and disadvantages. For the basic support structure of the spacecraft, primarily because of the emphasis on simplicity, reliability, and low cost, an aluminum-lithium alloy will be used. Composites, while having low density and very good strength per density ratio, are relatively expensive and have many potential problems with breakdown of the molecular structure under ultraviolet radiation and outgasing (molecules literally floating out of the matrix) in vacuum conditions. Composites also have poorly-understood thermal properties that are often unexpected and undesired.

Options	Advantages	Disadvantages
Conventional Al Alloys	• low cost • tried and proved • corrosion resistant • good strength/weight	
Newer Al Alloys (e.g. Al-Li)	• density 15% lower than Al • elasticity 20% higher • good strength & stiffness • cost dropping	• Cost slightly higher than conventional Al alloys • new & yet unproven
Superalloys (e.g. niconel)	• high temp. capabilities • very strong	• high cost
Titanium Alloys (e.g. Ti-Al-Mg)	• good strength/weight • very strong • high melting point • excellent corrosion resist	• somewhat expensive
Beryllium & its alloys	• good stiffness/weight • high melting point • lower density than Ti & Al	• relatively brittle • toxic (special NASA approval required)
Metal Matrix Composites (graphite/Al, graphite/Mg)	• good strength/weight • good crack resistance • good impact strength • low coeff. thermal expan.	• relatively new and untried • (currently) high cost
Composites (CFRP, GFRP)	• low density • good strength/weight • good stiffness/weight • can reduce parts count	• moderately high cost • difficult manufacturing • possible weakness to UV and radiation • outgassing under vacuum

Figure 8.8: Materials advantages & disadvantages

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As far as other metals, titanium is light, very strong, and has excellent corrosion resistance, but it is also expensive and in somewhat limited quantity. Beryllium, while used in small amounts in many spacecraft, especially in reentry shields, is quite brittle and is very toxic (special NASA approval is required for its use).

The conventional aluminum alloys are very well proven, over and over again. Aluminum-lithium is an alloy that, while lighter than most aluminum alloys by fifteen percent, it has an elasticity twenty percent higher. At the present time, Al-Li is a bit more expensive than conventional aluminum, but should be dropping in price as its use spreads.

Aluminum can not be used in many of the fuel tanks because of corrosive propellants. The main boosters (two Agena engines) burn "red fuming nitric acid" as an oxidizer, which is highly corrosive, with unsymmetric-dimethylhydrazine (UDMH) as the fuel. The oxidizer tank therefore must be made of stainless steel (or other more expensive materials), and the fuel tank of stainless steel or 1100 or 3003 series of aluminum.

Hydrazine (N_2H_4) and/or UDMH ($(CH_3)_2\ NNO_2$) are the fuels for the remaining boosters and all the thrusters. Therefore, all of the thrusters (and thruster fuel lines) and the remaining booster tanks must also be of stainless steel or aluminum 1100 or 3003. The oxidizer for the remaining two boosters is nitrogen tetroxide (N_2O_4), which is, if kept pure from moisture, stable and only mildly corrosive; thus the oxidizer tanks of the satellite booster and circularization engines can be aluminum.

Thermal Control Considerations

Thermal control is crucial in spacecraft that carry propellants, such as

MARTIAN does. The nitric acid oxidizer, for example, freezes below 231 K. As was explained above in PPS, space is normally at 3 K, but a black body at Earth will remain at 392 K and at Mars, 317 K. Thus some sort of thermal control is necessary. Passive thermal control is a system that attempts to radiate heat away/retain heat by simple radiation emission, heat conduction and/or heat convection, usually without turning anything on and off. An active control would require added sensors, actuators, and "heat pump". Thus a passive system, for example, reflecting energy by covering the craft with reflective aluminum foil, would be simpler and less costly. But on the other hand, an active system might be more effective.

One option is to paint the fuel tanks (and anything else that may need to be heated) with black paint with a high absorption constant. A problem with that is that even though the effective sunlit temperature at Mars might be acceptable, the same component might overheat at Earth. An active system could overcome this difficulty by sensing the temperatures continuously. But a problem for MARTIAN with active thermal control is the shear size of the vehicle. The larger the vehicle is, the more complex the active thermal control would have to be.

Problems, Concerns, and Needed Future Study

This design, being only the first response to a request for proposal, is only preliminary. There must be much more research in various areas before MARTIAN could become reality. A number of these areas involve the STR subsystem.

"Thermal control considerations" must lead to a specific thermal control system, capable of handling both the grueling cold of space and the relatively high sunlit temperatures, and yet keep the various components of the spacecraft at the desired temperatures.

Another consideration is the layout of the spacecraft. For this preliminary design, some assumptions were made, specifically on the modelling of unsymmetric bodies as symmetric in order to simplify the calculations of inertia, mass, and center of mass. A more detailed analysis must be run before continuing with the design.

The structure of the spacecraft has had little to test it. A full structural (static and dynamic) analysis must be done, either through analytical methods or through the use of a finite element program. The modifications done to AEROB were designed to add to the output a highest and lowest value for pressure on the aeroshield during aerobraking. This is one of the many forces that must be analyzed; the others are primarily caused by launch loads and loads when burning boosters.

The availability of the Heavy Lift Vehicle, although likely, remains to be seen. The use of shuttle would cause a great change in the design of MARTIAN, since it would have to be cut into several smaller pieces.

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Conclusion

We, the members of design group #6, believe that our design of the M.A.R.T.I.A.N. spacecraft has incorporated into it many innovative features that will enhance our mission's reliability and help to reduce it's overall cost. The most important of these design features are:

- •Flexibility built-in
  - -required to accomidate 2 different aircraft designs
- •HLV launches
  - -lower cost
  - -less on orbit assembly
- •Satellite orbit allows for
  - -constant communication with base/plane
  - -solar arrays in sunlight for 90% of orbit
  - -good sceince coverage of planet surface
- •Satellite kicked out during aerobrake maneuver
  - -lowers ΔV for satellite to reach orbit
  - -lowered propelant mass for satellite
- •TOCD improves mission reliability
- •Gravity gradient boom saves fuel for AACS
- •Aerobrake gives 30% mass savings over all propulsive

NOMADS

Nifty
Orbital
Martian
Aerobraking
Delivery
System

CRP 7
Incorporated

PERSONNEL LIST

Name	Social Security Number	Subsystem
James C. Lassa	[redacted]	Mission Planning
Charles W. White, Jr.	[redacted]	Mission Planning
Randall S. Weakly	[redacted]	Attitude/Articulation
Gregory P. Lehmann	[redacted]	Aerobraking
Jerrold Petrizzo	[redacted]	Structures
Russell A. Delaney	[redacted]	Command/Data Control
Richard C. Christian	[redacted]	Power/Propulsion
Laura J. Vanerka	[redacted]	Science

Table Of Contents

ABSTRACT

A conceptual design for a spacecraft delivery system has been developed to deliver the components of a manned aircraft to the Martian surface. The system design includes seven major subsystems, each integrated to satisfy individual and overall mission requirements. Numerical data is presented within each subsystem to support compliance with mission requirements. The overall delivery system design stresses simplicity, reliability, and low cost.

Introduction

The question when it comes to Mars is no longer how, but when. Soon after man arrives on Mars, a means of transporting men and material to distant sites of interest will be required. To fulfill this need, a manned Mars aircraft and the spacecraft system to deliver it have been proposed. This report is given in response to the Request for Proposal for the spacecraft system required to deliver the components of the manned aircraft to the Martian surface. A thorough preliminary design study has been conducted to determine major design concerns, establish the size of, define the subsystems for and describe the operation of the delivery system according to the following requirements:

1. The spacecraft will consist of two primary components: the payload reentry system and an instrument bus carrying scientific instruments for remote sensing of the planet's surface. The instrument bus will remain in orbit after separation from the reentry system.

2. The following subsystems are identified for the purposes of system integration:
   a. Aerobrake (including orbit capture, reentry, and detachment)
   b. Structure (including materials, design, and thermal control)
   c. Power and Propulsion
   d. Attitude and Articulation Control
   e. Command and Data Control
   f. Science and Radio Relay Instrumentation
   g. Mission Management, Planning and Costing

3. The spacecraft's components and payload will be delivered to orbit in the cargo bay of the space shuttle and be assembled on-orbit at the space station spacecraft assembly-and-repair facility. The extent of shuttle support should be identified and minimized.

4. The spacecraft will be able to be retrieved by a remote manipulation device on the space

station or space shuttle.

5. Nothing in the spacecraft's design should preclude it from performing several possible missions, carrying vastly different payloads to different destinations.

6. The spacecraft will have a design lifetime of four years, but nothing in its design should preclude it from exceeding this lifetime.

7. The spacecraft will use the latest advances in artificial intelligence where applicable to enhance mission reliability and reduce mission costs.

8. The design will stress simplicity, reliability, and low cost.

9. For cost estimating and overall planning, it will be assumed that four space delivery systems will be built. Three will be flight-ready, while the fourth will remain on the ground for use in an integrated ground test system.

10. Mission science objectivces are outlined explicitly within the Science subsystem.

This report consists of thorough descriptions of the design procedures and final design of the seven major subsystems outlined above.

Aircraft Interface Data

For purposes of successful aircraft integration, the following aircraft system interface information was made available by group three of the aircraft design section:

Total Mass: 1276.3 kg

Dimensions: 11 m long x 4 m (dia)

Range: 900 km

Configuration: unassembled (see Fig. 1)

No need to transport fuel.

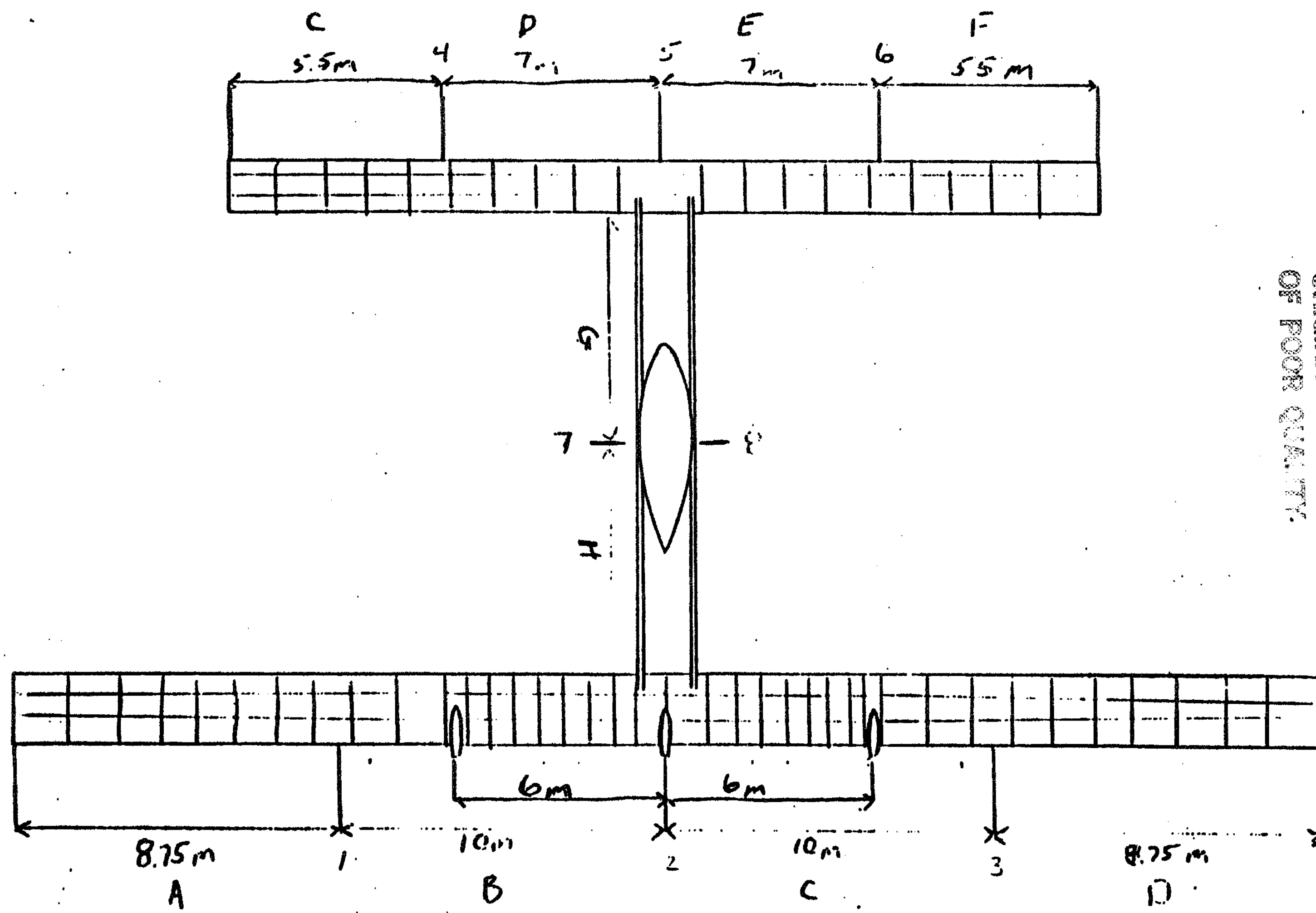

Fig. 1

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I. MISSION PLANNING

INTRODUCTION

Here's a look at the requirements for the Mission Planning subsystem:

- select target
- compute delta v to target
- design spacecraft to carry out mission
- integrate, test and launch spacecraft
- mission support
- end of mission, end of $

Figure 1.1 is a flowchart summarizing different options looked at, and providing an overall look at method of attack.

To further describe it, the mission has been broken down into four phases. The phases are defined as follows:

PHASE 1: Launch, cruise and orbit capture.

PHASE 2: Final orbit selection.

PHASE 3: Aircraft payload reentry.

PHASE 4: Orbit adjustment for aircraft communication support

and science reconnaisance.

The end of the mission is defined at the completion of phase 4.

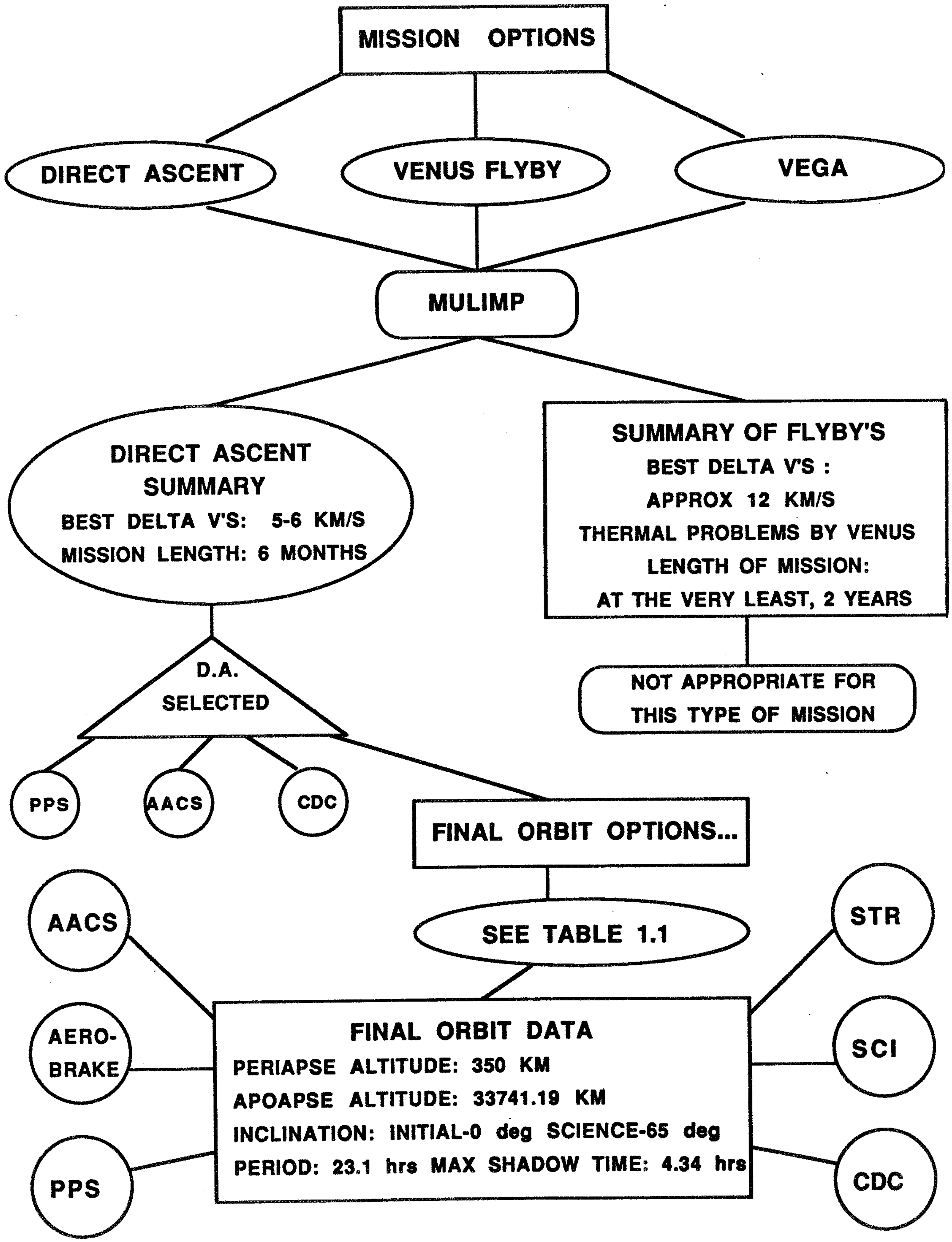
MISSION OPTIONS
DIRECT ASCENT
VENUS FLYBY
VEGA
MULIMP
DIRECT ASCENT SUMMARY
BEST DELTA V'S: 5-6 KM/S
MISSION LENGTH: 6 MONTHS
SUMMARY OF FLYBY'S
BEST DELTA V'S :
APPROX 12 KM/S
THERMAL PROBLEMS BY VENUS
LENGTH OF MISSION:
AT THE VERY LEAST, 2 YEARS
D.A. SELECTED
NOT APPROPRIATE FOR THIS TYPE OF MISSION
PPS
AACS
CDC
FINAL ORBIT OPTIONS...
SEE TABLE 1.1
AACS
AERO-BRAKE
PPS
STR
SCI
CDC
FINAL ORBIT DATA
PERIAPSE ALTITUDE: 350 KM
APOAPSE ALTITUDE: 33741.19 KM
INCLINATION: INITIAL-0 deg SCIENCE-65 deg
PERIOD: 23.1 hrs MAX SHADOW TIME: 4.34 hrs

Fig. 1.1

Choosing Delta V

The delta V choice was dependent on the type of mission selected. To satisfy the requirement for optimal delta V, a number of mission options were considered, mainly the three that follow: a direct ascent path, a trajectory including a Venus flyby, and a Venus-Earth gravity assist, or VEGA.

Phase 1

After taking a look at mission complexity, with the help of the delta V optimization program MULIMP (ref. 5), it was determined that flyby missions were not appropriate due to longer length-of-mission times (which led to higher delta v's), and thermal problems that would be encountered when flying by Venus. Therefore, the direct ascent trajectory was selected and, with the help of MULIMP once again, delta v was optimized with respect to launch date and total flight time. The results are plotted in Fig. 1.2. From Fig 1.2 it can be seen that an optimal delta v is approached approximately at instances when one synodic period has passed with respect to original planet geometry. Thus there is almost an endless number of launch windows corresponding to the delta v chosen for this mission, which is equal to 5.671 km/s. It can also be seen that optimal flight time corresponds closely to optimum delta v.

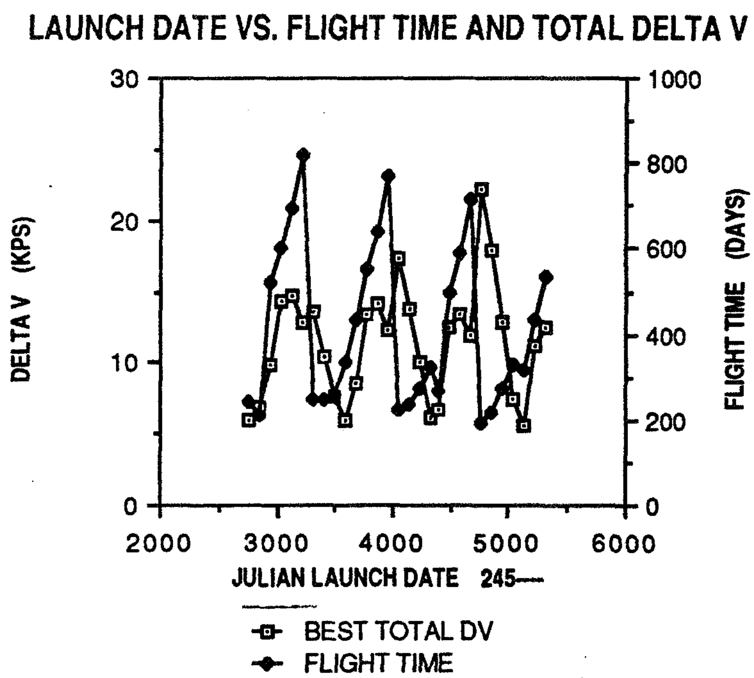

Fig. 1.2

Original planet geometry , including planet positions at launch and encounter, is detailed in Fig. 1.3. For a particular launch date Fig 1.4 gives the arrival delta v which is used to find the fuel requirement needed at launch and gives an indication of how long aerobraking will take and how much fuel it will use. The information provided in these figures also enable the power and propulsion subsystem to size fuel tanks and batteries accordingly, and also provide criteria for antenna sizing which is handled by the communications subsystem. The total spacecraft travel time, then, is approximately 204 for the transfer orbit and approximately 40 days (12 orbits) for orbit capture and aerobraking. See Section III for details on aerobraking.

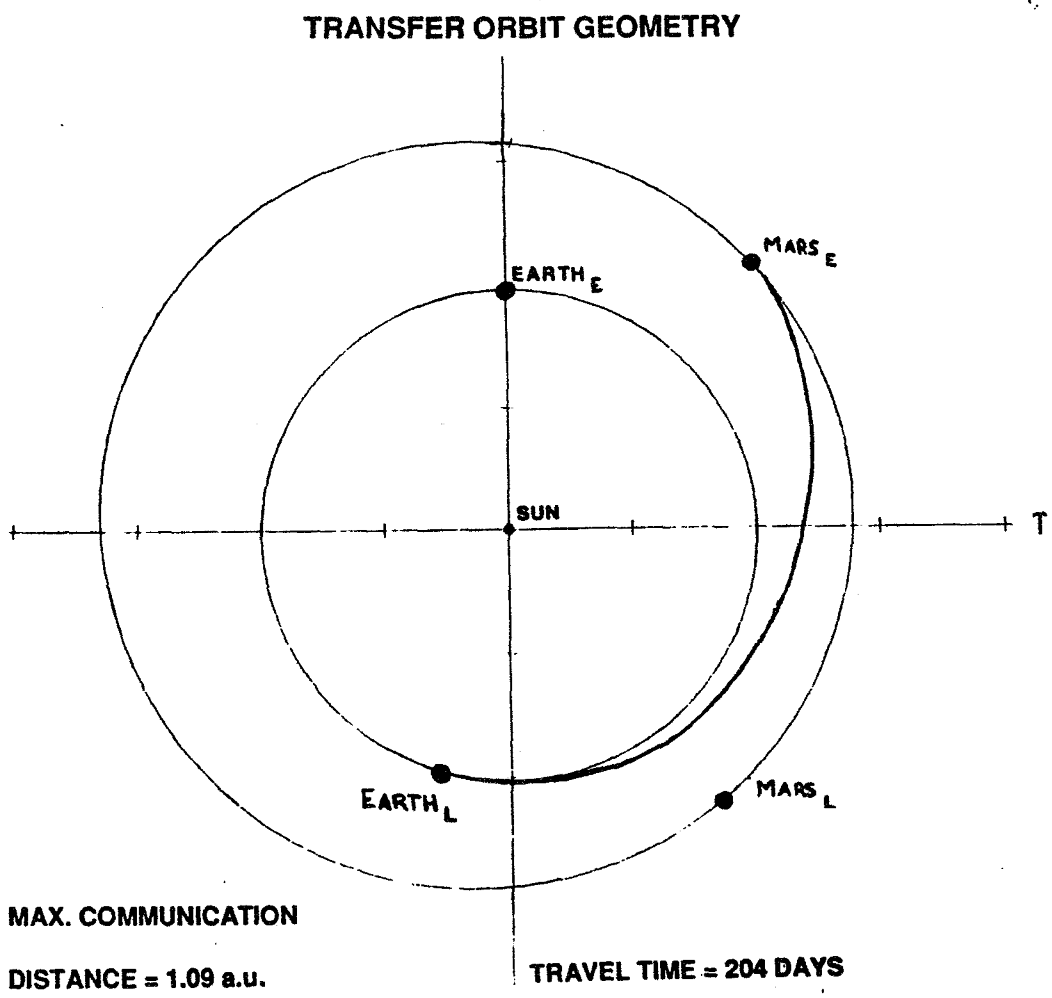

Fig. 1.3

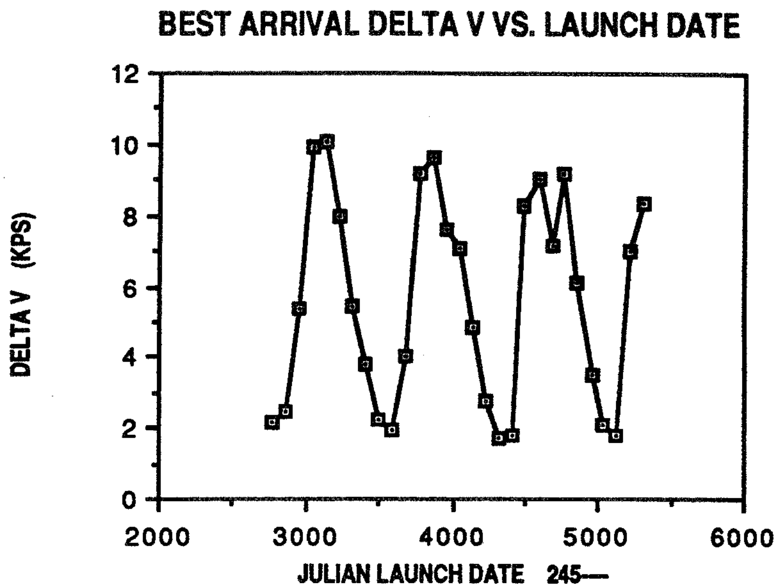

Fig. 1.4

Final Orbit Selection

In selecting the final orbit, we enter phases 2, 3, and 4 of the mission. Phase 2 concerns the type of orbit chosen at the completion of aerobraking (discussed in detail in the aerobrake subsystem analysis), while phase 3 brings the mission through payload reentry. Adjusting the orbit to its final configuration is phase 4.

Phase 2

To aid in the selection of a final orbit, a table was set up weighing different orbit possibilities against applicable mission requirements. The instrument bus that remains in orbit must act as a communication satellite to support the aircraft during flight time, and is responsible for providing adequate viewing range for the science instruments affixed to the bus. Shadow time encountered would effect the power subsystem, so a minimum eclipse mode was important. The time in eclipse mode directly affected battery as well as solar array sizing. The type of orbit selected also affected the amount of delta v spent during aerobraking. Finally, an orbit suitable for successful landing sight observation prior to committing for reentry was a top priority. The results of trade-off orbit comparisons are summarized in table 1.1.

orbit type → / objective ↓	circular equatorial synchron.	low (eq) altitude circular	elliptical equatorial synchron.	elliptical equatorial non-sync	elliptical inclined synchron.	elliptical inclined non-sync
communications with A/C	4	1.5	3	3.5	3	3.5
SCIENCE VIEWING	1	3	1	1.5	2	4
POWER REQ'S	4	2	3	3.5	3	3.5
landing the A/C payload	1	3	3	4	1	2.5
aerobraking, DV	2	1	4	3.5	3	2.5
TOTAL	12	10.5	14	16	12	16

Table 1.1

1= POOR 2= PASSABLE 3= PRETTY GOOD 4= EXCELLENT

The totals, after ranking how well each objective is satisfied for each of the various orbits, yielded a tie. Comparing the two orbits revealed that each had poor ratings in different pertinent objective categories, that of science reconnaisance and aircraft delivery. The similarities between each particular orbit led to the decision that each one could be used for a different phase of the mission and satisfy applicable requirements without radically increasing the total delta V required.

With the general orbit types selected, it was possible to assign specific values to the orbit parameters involved, calculating them to satisfy requirements for each particular phase.

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Phase 3

In order to view the condition of the predesignated landing sight, the science instruments need to be within approximately a 350 km altitude. This was the number chosen for the orbit altitude at time of periapse passage. A non-synchronous orbit had already been decided upon due to science requirements so the period of the orbit was selected to best comply with both science and landing objectives. A synchronous orbit would have a period of 24.6 hours. A period of 23 hours was chosen to precess the orbit 22.5 degrees per day. This will enable future science requirement compliance while still allowing adequate time for landing site reconnaisance. Given the period of the orbit and the required periapse altitude, the semi-major axis and the apoapse altitude can be found. The resulting orbit parameters are:

a : 19565 km
e : .809
period : 23.0 hours
periapse radius : 3730 km
apoapse radius : 35400 km

To check satellite time in eclipse mode (needed for power requirements) a graphical study was carried out to find the maximum amount of time that would be spent in a shadow. Using Kepler's equation, (1.1), and the intersection of the orbit and shadow, it was found that a maximum shadow time of 4 hrs 20 min (17 percent of the orbit period) occurs when the orbit is in the ecliptic plane.

After the completion of aerobraking and reconnaisance of the landing site, a Hohmann transfer orbit will be employed for payload reentry. The delta v required to insert the payload into a reentry trajectory is given by Eqn. (1.2) and is found to be 0.021 km/s.

$$\Delta v = \sqrt{2\,[\,\mu/r - \mu/(2a_1)]} - \sqrt{2\,[\mu/r - \mu/(2a_2)]} \qquad (1.2)$$

where a_1 is the semi-major axis of the original orbit and a_2 is that of the transfer orbit. Figure (1.5) summarizes these results.

Figure 1.6 takes a look at this final orbit configuration throughout the Martian year with respect to the ecliptic plane. From the figure, it can be seen that throughout most of the year, the satellite is nearly always illuminated. Thus, the time spent in eclipse mode is much less than 17 percent of the time, which means that no adjustments in solar array or battery size are necessary.

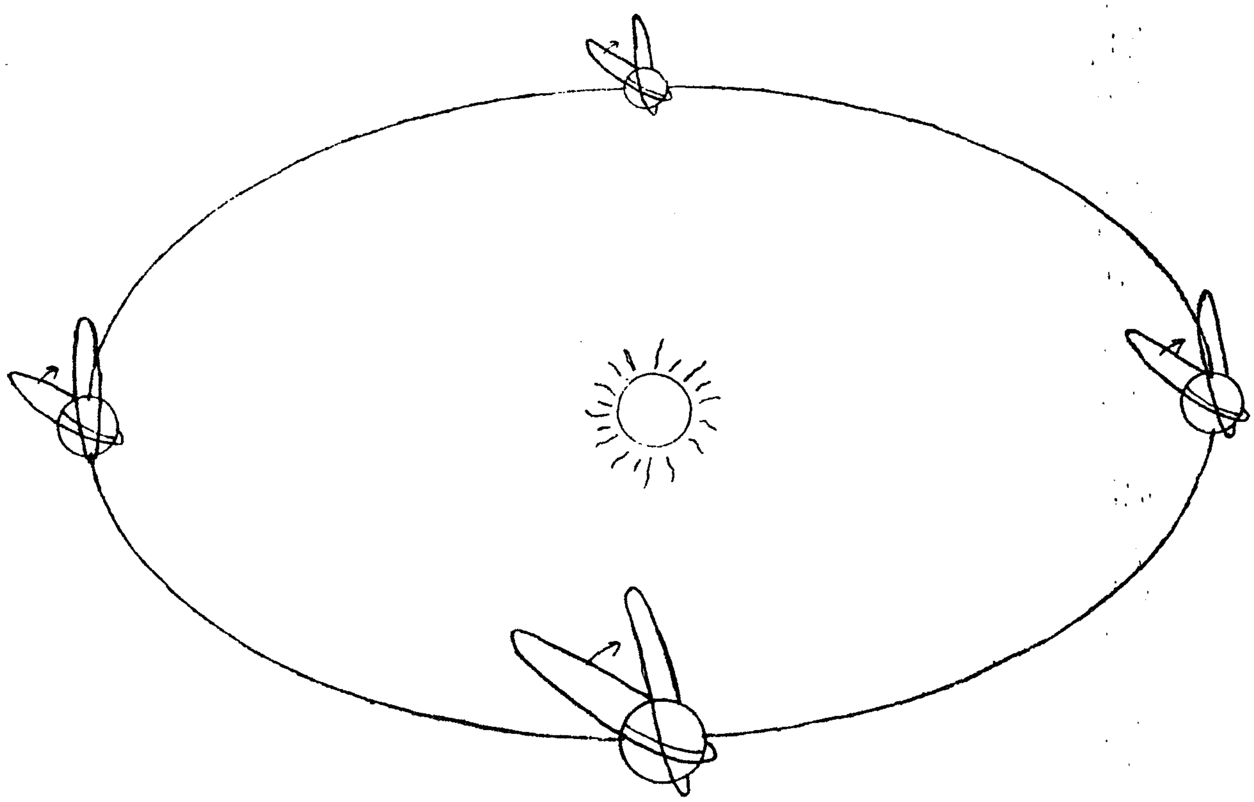

Fig. 1.6

Figure 1.7 shows how the aircraft commumication support requirement is met. The figure demonstrates where and when the satellite appears relative to the operating area. Part A in the diagram represents about 15 hours of the total orbit during which there are direct line of sight communication capabilities. The communication distance from aircraft to satellite in this part of the orbit ranges from 20,000 km to 32,500 km. Part B represents about 43 minutes during which there are line of sight communications possible between aircraft and spacecraft with an average range of about 350 km. Parts C and D depict the remaining 8 hours of the orbit when the satellite is below the horizon and aircraft to spacecraft communications are not possible. Finally, fig. 1.7 illustrates the angle of the satellite above the horizon.

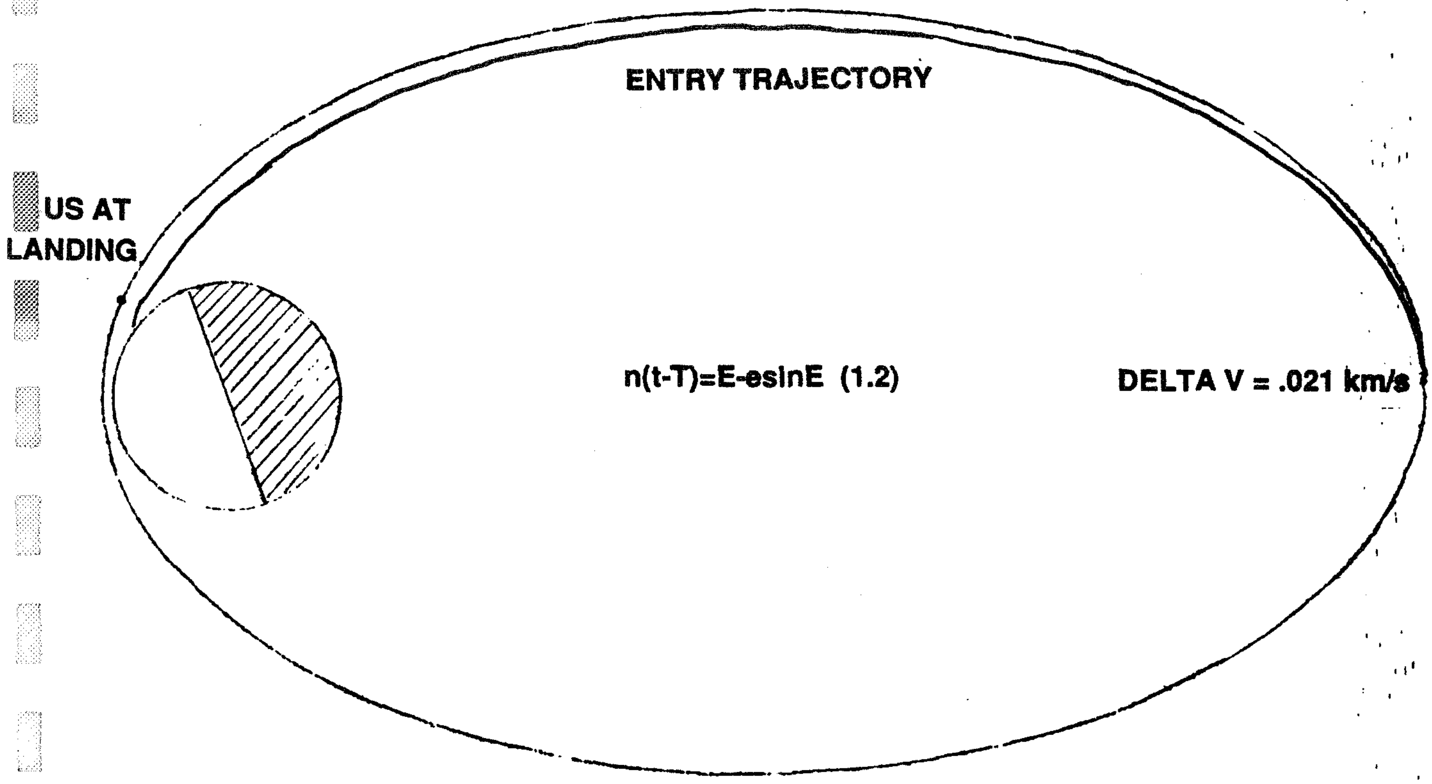

Fig. 1.5

Phase 4

With the completion of the successful payload landing, it is then possible to move into phase 4 of the mission, adjusting the orbit to meet the remaining science requirements while fulfilling the requirement for providing communication support to the aircraft. The orbit adjustment necessary is a change in inclination of 65 degrees. This will bring the polar caps into the range of the science instruments, a necessary step towards fulfilling science requirements. To minimize the delta v required for orbit adjustment, a burn maneuver is executed at apoapse, changing inclination but keeping the same period, perigee, and apogee. (For minimum delta v, the burn must occur where the orbit speed is smallest, which occurs at apoapse.) Thet delta v is given by Eqn. (1.3) and is found to be .517 km/s.

$$\Delta v = 2 v \sin (\Theta/2) \tag{1.3}$$

where v is velocity at burn point and Θ is desired change in inclination.

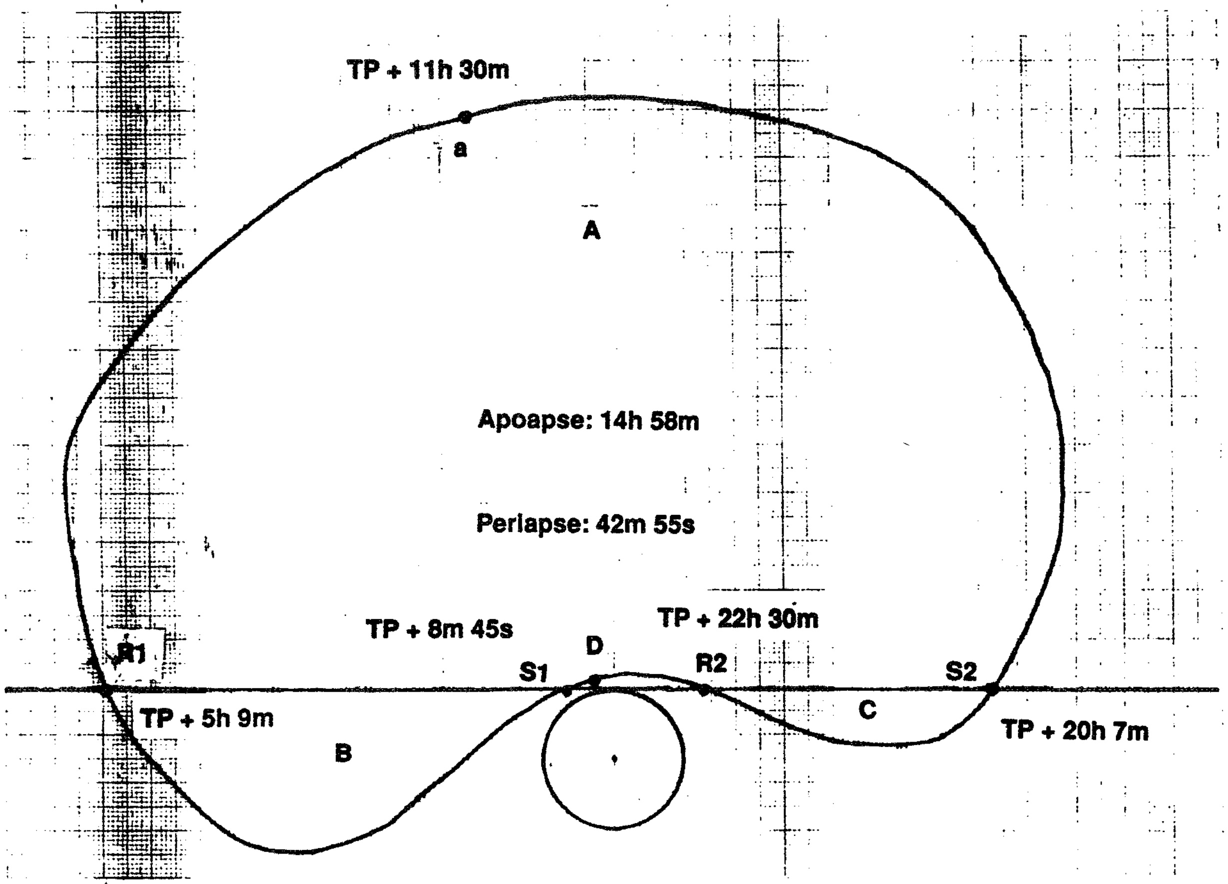

Fig. 1.7

Massive Considerations

The different subsystem masses are compiled in Appendix A, with a total (fuel, payload and all subsystem masses) spacecraft mass of about 27,650 kg. Since the Space Shuttle can carry 29,480 kg into low earth orbit, the weight factor is no problem. Due to the sizes of the various components, however, the whole package will not fit into one cargo bay, so more than one shuttle transport will be necessary.

The dimensions of the shuttle cargo bay are illustrated in Fig. (1.8). As mentioned earlier, the aircraft dimensions alone are 11m long x 4 m (dia.). Dimensions for the remaining components of the spacecraft delivery system to be transported to the space station on-orbit assembly area are as follows (approximated for the volume they will occupy):

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- Instrument Bus Satellite : 2m x 2m x 1m
  (with panels folded)

- Landing vehicle : 4.3m (dia.) x 11.3m

- Landing gear : 2 - 3.5m x 4m

- Aerobrake : 4 pieces, 5.5m x 7m x 5.5 m each

- Fuel tanks : 2 - 8m x 2m (dia.)
  1 - 6m x 2m (dia.)
  3 - 1.5m (dia.) spheres

Therefore, due to the volume restrictions of the shuttle cargo bay, at least two flights for transport would be required. A more detailed description of the component arrangement is covered in the Structures subsystem section of the report.

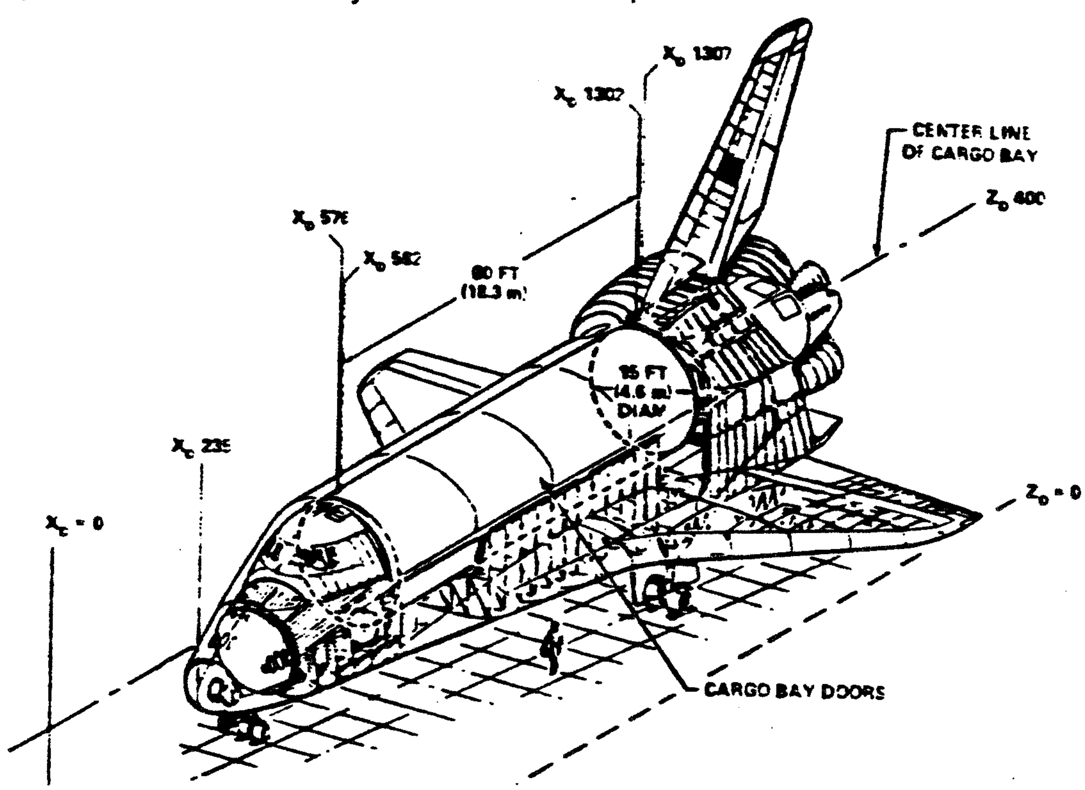

Fig. 1.8

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Cost Analysis

A cost analysis program was used to determine an approximate expense for the spacecraft delivery system. The program uses a mass-based model where total cost is a strong function of total mass. The results of the estimate are shown in Table 1.2.

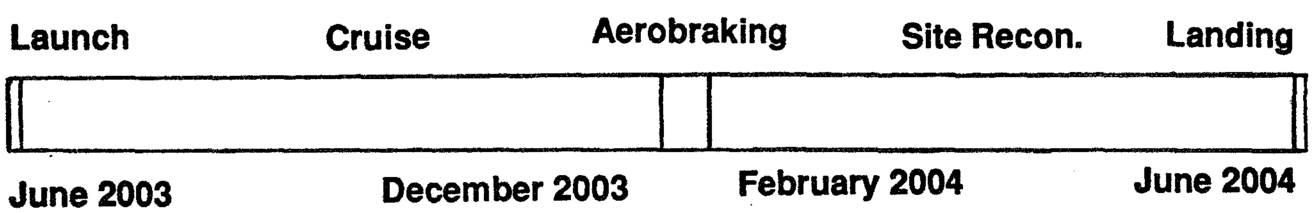
To summarize, since four delivery systems are to be built, the entire project has an estimated price tag of $6.4 billion. This is a very generous estimate and includes the cost of two shuttle flights.

Table 1.2

SUBJECT	D D T & E in mil of $	F H A in mil of $	TOTAL in mil of $
Structures	91.6	42.7	34.3
Thermal(AB)	23.5	39.8	63.3
Attitude Control	162.4	37.8	200.2
Articulation Control	69.3	8.2	77.5
Commun-ications	64.3	13.9	78.3
Power	11.7	1.0	12.7
Propulsion	2.0	0.0	2.0
System test Hardware	326.0	----	326.0
System test Ops.	46.0	----	46.0
Software	0.0	----	0.0
GSE	43.5	----	43.5
S E & I	93.5	21.0	114.4
Program Mngt.	56.4	20.8	77.2
Contingency	198.2	37.0	235.2
Fee	118.9	22.0	141.1
Program Support	26.2	4.9	31.1
TOTALS	1334.1	249.4	1583.5

Implementation Plan & Mission Timeline

January 1988 - May 1988	Preliminary Design Study
June 1988	Acceptance of complete contract
July 1988 - December 1989	Final Design Completion
January 1990 - December 1995	Development and Testing
January 1996 - December 1997	Completion of Ground Based Test System
January 1998 - December 1999	Completion of individual components of First Launch-Ready Delivery system and continued On-Going tesing with ground based system
January 2000	First of two shuttle flights carrying components of delivery system to space station to begin assembly
February 2000	Second Flight (including Martian aircraft)
December 2001	Completion of third system
February 2002	Assembly of instrument bus and lander complete
March 2003	Assembly of Aerobrake complete
June 8, 2003	LAUNCH of first system(barring any set-backs)
December 2003	Completion of final system



Problem Areas

1. Not being able to perfectly satisfy Mission Science requirements and all-time aircraft communications support simultaneously due to having only one satellite in orbit.

Solution: Install a separate satellite for the sole purpose of communication, the next logical step.

2. Not being able to meet proposed launch date due to unexpected design, development, testing, or political problems.

Solution: Acceptable launch windows appear once every Earth-Mars synodic period (about 2.2 years).

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II A. Propulsion subsystem

In order for our spacecraft to deliver its payload successfully to the surface of Mars, a propulsion system is needed that will do the job safely and effectively. In this section of the report, several propulsion systems are analyzed. Selecting the best system for this mission begins by determining the propulsive requirements as they are specified in the RFP. The propulsion system chosen will be the one that meets these requirements to the greatest degree. They are as follows:

1.) PPS will deliver the spacecraft payload to a pre-determined Martian orbit, and finally the planet's surface.

2.) It will use off-the-shelf hardware where available.

3.) Its design is restricted from using materials or technology expected to be available for production only after 1998.

4.) The design stresses simplicity, reliability, low cost, and minimum mass.

There are several considerations to be taken into account when designing a propulsion system, and also several types of systems to study. After researching this field, five propulsion systems were studied and evaluated. They are as follows:

1.) solid chemical propulsion

2.) liquid chemical propulsion

3.) nuclear fission propulsion

4.) electric ion propulsion

5.) solar/laser thermal propulsion

The advantages and disadvantages of using these different types of systems may be found in Table 2a.1.

Because of the additional burns required for aerobraking when the spacecraft arrives at Mars, solid chemical rockets are discarded. The need for simplicity dictates that electric ion and nuclear fission propulsion also be discarded, although both of these fields are currently being researched, showing promising hope for these types of systems to be effective in the near but not immediate future. Solar/Laser thermal propulsion systems are also discarded due to the need for very high laser pointing accuracy limiting them to only near-Earth missions.

All things considered, a liquid chemical propulsion system is indeed optimal for this mission. The state-of-the-art liquid chemical engines perform with an Isp of 460 seconds, providing high thrust, as well as, reliability and relative simplicity. These engines can also easily be stopped and restarted for additional delta v maneuvers, as will be needed when the spacecraft reaches Mars. In addition, liquid chemical technology is currently available, and off-the-shelf hardware will be implemented wherever possible. Even though a higher relative mass is obtained with this type of propulsive system, and complexity is increased by the need for a liquid hydrogen refrigeration system, liquid chemical rocketry successfully meets the specified design requirements listed earlier.

Table 2a.1. Propulsion system advantages and disadvantages

system type	advantages	disadvantages
1.solid chemical	1.simple and reliable 2.high density Isp allowing for small system 3.very stable propellant therefore very safe 4.high thrust 5.ease of storage	1.low Isp (200-300) seconds 2.no refuelling capability 3.non-restartable 4.use only for "single shot missions"
2.liquid chemical	1.high Isp (370-500) sec 2.easily restartable 3.refuellable for reuse 4.H2O2 engine exhaust is non-toxic (water) therefore safe testing	1.more complex than solids 2.more massive 3.less easily stored 4.cooling system required
3.nuclear fission	1.high Isp (1300-1500)sec 2.high stage mass fraction	1.political question of nuclear radiation 2.large, complex system 3.public resistance to testing
4.electroc ion	1.higher Isp (1000-5000) seconds 2.low mass 3.easily stored	1.low thrust system resulting in very long mission times 2.large stage weight 3.much greater complexity
5.solar/laser thermal	1.moderate thrust 2.high Isp (1800-1200) sec	1.large solar collectors needed for solar powered vehicle 2.high laser pointing accuracy needed limiting missions only near Earth.

Liquid chemical propulsion system design

In designing the liquid chemical propulsion system for our Mars mission, liquid hydrogen-liquid oxygen fuel is selected because it represents the state-of-the-art in liquid chemical rocketry. It has an Isp of 460 seconds and has been proven very reliable in the past.

For this Mars mission, two different types of fuel systems are needed. The first of these is the LH2/LO2 system. It provides the initial thrust from low Earth orbit (LEO) to the Martian atmosphere. Aerobraking maneuvers are then performed, and additional delta v burns at this point require additional propellant from these tanks. Also, a re-entry burn must be executed after separation from the scientific instrument satellite, taxing the LH2/LO2 tanks for a final time.

The second system needed meets the requirements of attitude control and final descent thrusting. This system uses a monopropellant of liquid hydrazine (N2H4)which has a lower Isp (255 seconds), but is very useful in applications such as these. Since only a single chemical is used, complexity of the fuel feed system is reduced, decreasing its weight and increasing reliability. Our hydrazine propulsive system has three distinst subsystems. They are as follows:

1.) N2H4 tanks for thrusters to be used in attitude control of the spacecraft on its way to Mars. These thrusters and tanks will also be used for payload stabilization upon re-entering the Martian atmosphere.

2.) N2H4 tanks for final descent thrusting leading to a soft payload landing.

3.) N2H4 tanks to be used for attitude control of the scientific instrument satellite in orbit above Mars.

The propulsive requirements used in attitude control of the satellite will not be looked at here, but can be found in the AACS design/analysis section of this report. The two other hydrazine systems, as well as the liquid hydrogen-liquid oxygen propulsion system, will be presented here.

LH2/LO2 system design

There are several components to a LH2/LO2 system; the most important, perhaps, being the actual fuel tanks. For sizing of these tanks, factors to be taken into consideration are payload mass, other subsystem masses, the delta v requirements as specified by mission planning, tankage factors, fuel and oxidizer densities, and geometrical tank shapes, as well as space available for these tanks onboard the spacecraft. Payload and subsystem masses can be found in Appendix A, while the other factors mentioned above and obtained from other subsystem design analysts are located in Table 2a.2.

Table 2a.2. Requirements and needed constants for LH2/LO2 analysis

Delta v for initial thrust	5671 m/s
Delta v for aerobraking maneuvers	747.76 m/s
Delta v for re-entry burn	254 m/s
Total delta v	6672.76 m/s
Isp (LH2/LO2)	460 seconds
density of LO2	1141 kg/m3
density of LH2	70.79 kg/m3
g0=local gravity constant	9.8 m/s2
TF (tankage factor) for liquid chemical systems	0.15

The rocket equation is used initially to obtain the mass of propellant needed for the total delta v requirements mentioned above.

$$\text{delta v} = (g0)\ (Isp)\ \ln\ (Mi/Mf)$$

Result is that Mi/Mf = 4.39, where Mi is the initial mass of the spacecraft and Mf is its final mass after the propellant has been burned.

In the following equations, Mp is the mass of the propellant, 0.15Mp is the mass of the propellant tanks (a 15% TF), and 3095.7 is the mass of the rest of the spacecraft.

$$Mi = Mp+0.15Mp+3095.7$$
$$= 1.15\ Mp+3095.7$$
$$Mf = 0.15\ Mp+3095.7$$

Plugging these values into the equation above results in the following:

Mp = 21,351 kg = mass of propellant required

0.15 Mp = 3203 kg = mass of the propellant tanks

Using a mass ratio of oxidizer/fuel = 6/1, gives

mass of LO2 = 18,301 kg

mass of LH2 = 3,050 kg

Using the densities found in Table 2a.2, their volumes are as follows:

volume of LO2 = 16.0 m3

volume of LH2 = 43.0 m3

Now that the volumes of fuel and oxidizer have been obtained, proper tank sizing analysis can be executed. In the structures section of this report, diagrams show the layout of the aerobraking system and payload configuration. These constraints require the placement of two liquid hydrogen tanks and one liquid oxygen tank as depicted. Each tank has three parts. Each is comprised of a cylinder and two hemispheres. Working with a tankage factor of 15% and the formulas in Table 2a.4, the volumes, the shapes, and the masses of the tanks are calculated and then tabulated in Table 2a.5.

Fuel feed systems

There are two main types of fuel feed systems in liquid chemical rocket propulsion systems, i.e. pressure-fed and pump-fed. In the first type, high pressure tanks are used to hold the propellant, and helium is used as the pressurant gas. In a pump-fed system, propellant is transfered to the rocket engine by means of a pump and pumping aparatus. In this particular application,
a pressure-fed system was chosen to comply with the requirements of simplicity and reliability, eliminating the pumps and gearboxes which are prone to failure.

N2H4 (hydrazine) thruster design and configuration

In a monopropellant system such as the one implemented here, no oxidizer is needed; only fuel tank sizing must be carried out.

In the spacecraft attitude control thruster system, three small hydrazine tanks are needed for the thruster configuration as is shown in the diagrams found in the structural section of this report. Their sizes have been calculated as a result of fuel estimates given in Table 2a.3 and formulas found in Table 2a.4. The results for the AACS control of the spacecraft are tabulated in Table 2a.6.

A similar analysis is needed for the hydrazine (N2H4) tanks to be used during final descent of the payload. However, only one fuel tank is needed due to the payload landing vehicle's structural configuration. Details of the landing thruster's configurations may be found in the structural and aerobrake sections of this report. The results of the payload thruster tank sizing analysis can be found in Table 2a.7.

Table 2a.3 Requirements and needed constants for N2H4 analysis

mass of N2H4 needed for final descent to surface	90.8 kg
mass of N2H4 needed for attitude control of spacectaft en route to Mars and for control during re-entry	136.2 kg
Isp (N2H4)	225 seconds
density of N2H4 (hydrazine)	1008 kg/m3
g0=local gravity constant	9.8 m/s2
TF (tankage factor) for liquid chemical systems	0.15

Table 2a.4. volume formulas

shape	volume
cylinder	(3.142) (r2) (h)
sphere	(1.33) (3.142) (r3)

In each of the tank sizing result tables that follow, results are for unfuelled tanks. Masses of needed propellants can be found above and in Table 2a.3.

Table 2a.5 Results of LO2/LH2 tank sizing

number of tanks	type	mass (kg) per tank	dimensions (m)		volume (m3)	
					required	calculated
1	LO2	2745.2	r=1	h=4	16.0	16.7
2	LH2	228.8	r=1	h=4	43.0	46.0

Table 2a.6 Results of attitude control tank sizing

number of tanks	3
type of fuel	N2H4 (hydrazine)
mass per tank	6.81 kg
dimensions	sphere r = 0.32 m
volume per tank	0.14 m3

Table 2a.7 Results of payload thruster tank sizing

number of tanks	1
type of fuel	N2H4 (hydrazine)
mass per tank	13.62 kg
dimensions	sphere r=0.28 m
volume per tank	0.09 m3

Technical Problem Areas

A final note on the propulsion system designed here addresses some technical unsolved problems. The first of these concerns itself with the main rocket engine. Diagrams contained within this report depict only one main engine, however, during the production/testing phase of this mission it may be found that a configuration of more than one smaller engines would produce a better overall design. Regardless, placement of the engine(s) and proper structural considerations must be taken into account. A constraint here is that the rocket engine's thrust vector must pass through the spacecraft's center of mass. A gimballing angle may be needed to direct the rocket engine and orient it in a proper fashion.

Another problem to consider is concerned with the refrigeration system needed for the liquid hydrogen fuel. Although mass and power estimates have been made for this system, its placement and operation have not yet been determined. The liquid hydrogen, however, must be kept in the liquid phase for proper operation of the rocket engine.

A final problem that may need some looking into is proper sizing and placement of the pressurant gas tanks. In this report's diagrams three helium pressurant tanks are depicted. Their placement, as well as, their sizes and masses must be taken into account before production of our spacecraft can begin.

IIB. Power Subsystem

In order for the Mars aircraft mission to be successfully started and finally completed, all of the subsystems involved in the design must have their electrical power needs met. In this section of the report, power requirements are identified, and a power system is designed to meet these specified requirements. They are as follows:

1.) The power system (EPS) must be self-powered.

2.) It must send telemetry to C3 subsystem.

3.) It must accept commands from C3 subsystem.

4.) It must be able to sense temperatures, and sense power loads.

5.) It is required to control valve actuations and to control power relays.

6.) EPS will provide an uninterrupted source of power to all spacecraft loads during the mission life.

7.) It will protect the main power bus and power units against damage due to load faults.

8.) It must protect user loads against outages and damage due to EPS unit failures.

9.) EPS will control and process the power source (solar arrays) and the energy storage device outputs (batteries) into forms compatable with subsystem needs.

10.) It must provide commands and telemetry for EPS monitoring and control by mission support team.

11.) As always, the design must not use technology available only after 1998; it will use off-the-shelf hardware where available; its design will stress simplicity, reliability, minimum mass, and low cost.

Table 2b.1 lists the subsystem power needs as estimated by the design analysts.

Table 2b.1 power estimates

subsystem	SCI	C3	AACS	AEROB	POWER	PROPULSION
power load (watts)	86.2	130.0	180.5	160.0	150.0	78.4

Total power load = 785 watts

The first consideration in designing an electrical power system is to think about the different modes of power that the spacecraft will encounter throughout the mission. These are as follows:

Power modes

1.) Launch mode

During launch of the spacecraft, all subsystems must be prepowered. Thruster valves must be actuated, attitude control is needed from the start, and the central command computer must already be in operation. These initial power needs will be met through the use of an umbilical cable stretching from the orbitting space station to our spacecraft. It will charge the batteries that are installed aboard, providing the necessary power to all subsystem loads.

When the spacecraft is launched, the umbilical power cable will be detached, and solar arrays will be deployed for the cruise made, supplying needed power and charging the batteries.

2.) Cruise mode

As mentioned above, solar arrays are deployed in this mode, charging the batteries and meeting all subsystem power needs.

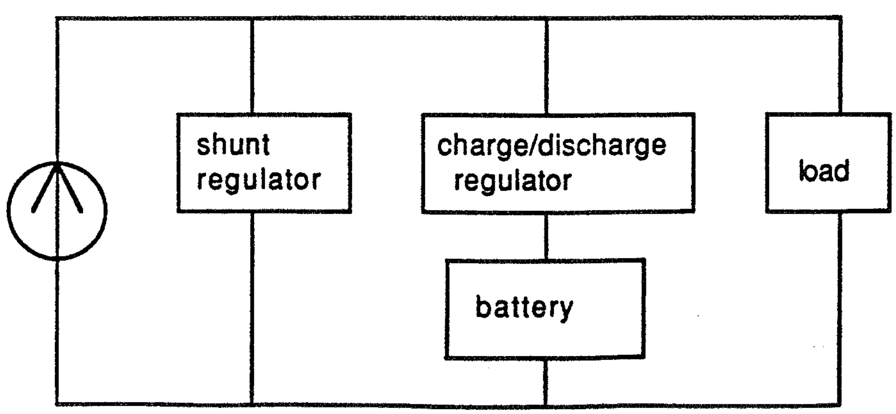
3.) Aerobraking maneuvers mode

When the spacecraft uses the Martian atmosphere for aerobraking to reach final orbit, the solar arrays on board will have to be retracted behind the aeroshell for protection.

4.) Separation mode

In this mode, the solar arrays and most of the battery cells will remain on the orbitting satellite to meet its specified power needs. A small number of batteries go down to the Martian surface with the aerobrake and payload to control attitude thruster valve actuations, as well as thruster valve actuations for the final landing procedure.

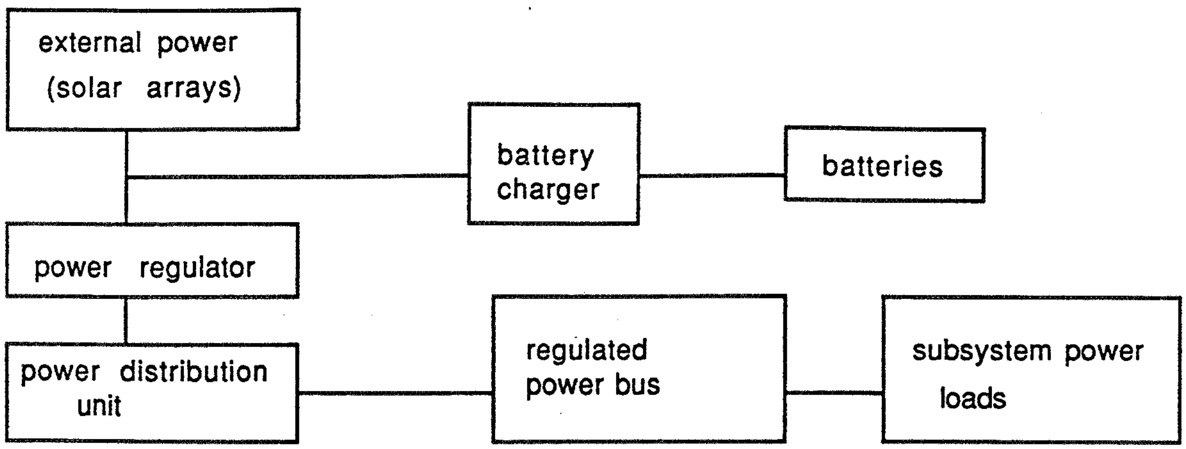
Fig. 2b.2 Regulated power bus



A regulated power bus is selected instead of an unregulated power bus, due to the fact that it is able to maintain a constant stable supply of voltage. Even though it has a higher mass and is more complex, the regulator will protect the user loads and the power bus as specified in the design requirements. A schematic diagram of the regulated power bus is found in figure 2b.2.

We want the electrical system to be designed so that there are no single point failures. That is, if any component were to fail, it would not cause the entire power system to fail along with it. EPS component failure is prevented by fusing individual battery cells or solar array strings. Load component failure is prevented by having parallel redundant fuses on each load, and a dual power bus prevents harness failure. Figure 2b.3 shows a diagram of the electriical power system simplified into its major components.

Fig. 2b.3 Power system



Power options

Power options to be considered for this mission are photovoltaic solar cells, solar dynamic power systems, and radio thermal generators (RTG's). The latter two were discarded and the former chosen for the following reasons:

1.) Photovoltaic solar arrays are vastly simpler than either solar dynamic systems or RTG's.

2.) Solar arrays meet subsystem power needs with minimal mass.

3.) Nuclear power in space is still of important social/political concern.

Solar array options

There are three major types of solar photovoltaic power: planar solar arrays, blanket-type solar arrays, and high concentrtion solar cells. Blanket-type solar arrays are the best option for this mission because of the need for folding retraction during re-entry into the Martian atmosphere. High concentration solar cells are still under development for efficient power production, planar solar arrays cannot be retracted, and blanket-type solar arrays fulfill the "off-the-shelf" power component requirement.

Battery option and selection

Along with the external source of power that solar arrays provide go rechargeable batteries to store additional power. These batteries will be used for both power storage and for power sources.

Two different types of rechargeable batteries were studied: nickel-cadmium and nickel-hydrogen. Nickel-cadmium batteries are selected for use on this mission for the following reasons:

1.) Ni-Cd's have twenty years of proven flight history.
2.) Missions of over seven years have been realized with Ni-Cd's.
(This fulfills the four year mission lifetime requirement.)
3.) Ni-Cd standard battery design already exists.
4.) Ni-H batteries are of unjustified higher cost.
5.) There is also difficulty in managing their electrolyte,
6.) Ni-H batteries must be pressurized; Ni-Cd batteries do not have this constraint.

Battery sizing analysis

Table 2b.4 contains information essential to the analysis of both batteries and solar arrays selected for this mission.

1.) number of battery cells needed =

(PL) (TE)/ (DOD) (watt-hours/cell) = 9.27 Therefore, ten battery cells are needed.

2.) battery capacity C = (PL) (TE) / (DOD) (V) = 8.47 ampere-hours.

3.) battery mass = (PL) (TE) / (DOD) (watt-hours/kg) = 11.86 kg

Therefore, each battery cell has a mass of 1.2 kg.

Solar array sizing analysis

1.) solar array power required (PBOL)

PBOL = (PT) / (time degradation)

PT is given by PT = PL + (C) (V) (DOD) / (TS)

Therefore, PBOL = (PL + (C) (V) (DOD) / (TS))/(time degradation)

PBOL = 1352 watts

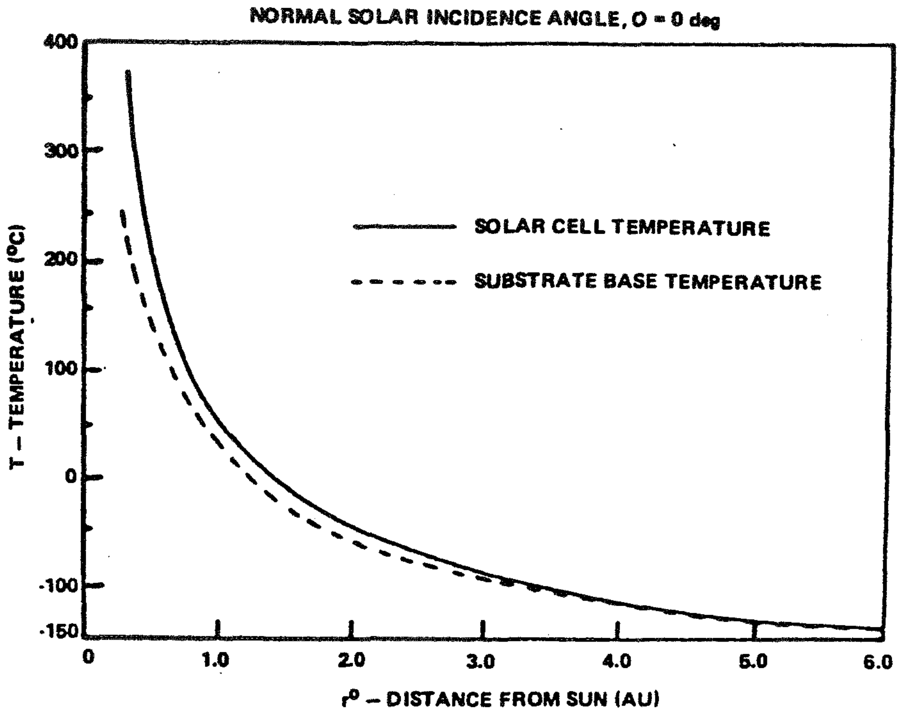
2.) solar array area

PBOL = S x Cr x e x A x (1-alpha (T-25))   area A = 24.4 m2

3.) solar array mass   mass = area x area density   mass = 38.8 kg

Table 2b.4 Battery/Solar array data

Total power load	785 watts
Mars orbit distance about the sun	1.5 A.U.
time in eclipse mode (TE) -obtained from mission planning	0.17 hours
time exposed to sun (TS) -obtained from mission planning	0.83 hours
nominal bus voltage (V)	35 volts
maximum depth of discharge (Ni-Cd)	45%
energy density (Ni-Cd @100% DOD)	25 watt-hours/kg
energy per battery cell	32 watt-hours/cell
solar cell effeciency (e) @ 25 degrees C -effeciency drop of 0.5% per degree C	12.0%
operating temperature of deployed array	50 degrees C
total degradation of solar cells (radiation, etc.)	30% in 5 years
solar constant @1.5 A.U. (Mars orbit) (S)	600 watts/m2
packing factor of solar cells	88%
area density of blanket-type solar arrays	1.59 kg/m2



Solar array configuration is shown in the structural subsysten section of this report. There are four solar arrays, each having the same dimensions and characteristics. Each array will be 6 meters long and 1 meter wide giving a total array area of 24.0 m2. This should be sufficient to supply enough power to the spacecraft, taking into account the fact that not all subsystems will be taxing the power batteries at the same time. Table 2b.5 shows the results of the battery and solar array sizing.

Table 2b.5 Battery/Solar array sizing results

Ni-Cd rechargeable batteries

number of batteries	mass per battery	total battery mass
10	1.186 kg	11.86 kg

Solar arrays

number of arrays	mass per array	total array mass	area per array	total array area	array dimensions
4	9.7 kg	38.8 kg	6.0 m2	24.0m2	1m X 6m

Re-entry battery analysis

As mentioned before, most of the batteries will remain in orbit giving needed power to the scientific instrument satellite, and some will go down to the Martian surface in order to power the re-entering subsystems. The aerobraking subsystem power requirement is equal to ((PL) (TE) /DOD)/(watt-hours/cell) = 1.89

Thus, two battery cells must be mounted within the aerobraking subsystem, leaving eight to remain inorbit on the satellite. The satellite power load is 625 watts. Therefore, n = (625) (0.17)/(0.45) (32) = 7.38. This corresponds, as predicted, to eight battery cells remaining in orbit. As a final note, to the propulsion and power subsystems, the following listing describes PPS interaction with other subsystems.

Interactions with other subsystems

1.)Science
- -scientific instruments power needs

2.)Attitude control system
- -computer and sensing devices power needs
- -additional propellant needed for attitude corrections during flight, in orbit, and upon re-entry

3.)Communications
- -central control computers power needs
- -power required for antenna pointing

4.)Aerobrake
- -additional propellant needed for re-entry burn and for retro-rockets when delivering payload to the planet's surface

5.)Thermal control (Structures)
- -propellant tanks are a major structural consideration
- -cooling system is needed for liquid hydrogen fuel

6.)Mission planning
- -initial boost delta v requirements

7.)Power and propulsion
- -must interact with itself for its own power requirements (deploying and retracting solar arrays, rocket valve actuations, etc.)

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AEROBRAKE SUBSYSTEM

Aerobraking is a technique which uses sucessive passes through the upper atmosphere to circularize a highly elliptical orbit. Strictly adhering to "successive passes" and the "upper atmosphere" keeps operations and technology simple. Using this aerodynamic drag to lower orbit energy, instead of a heavy retropropulsion system, results in significant reductions in propellent and structural mass, thus lowering costs.

On this mission, NOMADS (see table1 for requirements), on a trans-mars trajectory, travelling at 2.702 km/s is captured in a highly elliptic orbit of 100,000 km semi-major axis by a propulsion burn of 731.8 m/s delta V. Once in this elliptic orbit, a small apoapsis propulsion maneuver would lower the periapsis in to the upper atmosphere(fig 3.1). Raising the periapsis could stop braking at any time for surveying or avoiding trouble. At a semi-major axis of 19,390 km, the instrument bus (satellite) is separated from the entry vehicle. The entry vehicle then continues to aerobrake down to a nearly circular orbit. A small burn captures the atmosphere and the entry vehicle deorbits, decellerates under a parachute and descent engines, and softly lands.

A proposed drag device consists of a gas filled ballute which is deployed prior to the entry maneuver and surrounds the vehicle and the payload with exception of the main engine (fig 3.2). The vehicle is positioned so that the rocket faces forward. This ballute could represent a considerable increase in mission payload relative to the conventional baseline aerobrake system. The ballute concept developments should be looked into in order to assess possible use in the future. This is a viable alternative to the conventional aerobrake if advances in materials progress at a rapid rate. The 1998 deadline requirement on technology comes into the picture here. The ballute will probably not be tested and reliable by then.

aerobrake size and shape

A conventional aeroshield was selected because of its reliability and lower cost. The size was optimized using AEROB. A frontal area of 120 square meters is found to be optimal (fig3.3). The shape is chosen to be a blunted wide angle cone that serves as part of the structure of the stage. This particular cone is an elliptical cone raked off at an angle of 73 degrees. This cone was selected because it has a flatter shape for surface area efficiency and it provides lower stagnation point heating than sharper cones. The desired lift-to-drag of 0.3 requires a rake angle of 73 degrees and appears to be sufficient for control of the vehicle during atmospheric flight. Control in the aerodynamic braking part is accomplished by varying the "effective" lift-to-drag ratio (equation 3.1)[3.1]. The base of the cone is an ellipse with a length of 14m and a width of 11m. This configuration was chosen to more efficiently cover the long payload.

equation 3.1 $$K_{eff} = K_b \cos(a)$$

K = Lift to drag ratio

a = banking angle

layout of aerobrake

The layout of the vehicle was arrived at by balancing various requirements and considerations.

One gound rule for the study is that the vehicle be transportable in the Orbiter payload bay. This affects the development of the shield and how it must be broken down. Another demand is that the entry vehicle be of a stable configuration during entry into the Martian atmosphere. To insure stability, the center of gravity should be kept ahead of the meta-center in the range of angles of attack to be considered in this design. A technical problem is to find this meta center at each angle of attack. A testing system must be developed to determine these angles. The layout (see Structures) shows the payload located in a cylinder as it will be for the entire flight. It is protected from heating during aerobraking and entry. The most severe heating occurs at the neck of the wake coinciding with the rear stagnation point. In order to insure the payload is kept away from the neck, the payload should be kept within a length of two diameters of the shield. The heating in this region is less than 3% of the heating at the front stagnation point[3.2]. The payload and instrument bus are well within this constraint.

A tradeoff in the direction of the main engine determined the orientation of the payload, the instrument bus, and the tanks relative to the shield. Three possible firing arrangements are possible: side firing, firing through shield and firing away from the shield. The last firing arrangement would require the engine and tanks to be placed arround the outside of a central payload. Since the payload is long, exhaust from the main engine and entry heating would interfere with the payload and bus. Firing through the shield would be too complicated because doors would have to be built to allow firing in this manner. Therefore, a side firing arrangement was selected to avoid complexity and to conserve on the amount of structure that would be necessary to protect the latter arrangement.

Optimization of the final mass injected was determined using AEROB and optimizing the semi-major axis verses shield area as shown in figure 3.3. The optimal semi-major axis occured at 100,000km with a shield area of 120 square meters. The optimization process was acomplished by varying parameters such as maximum shield temperature, initial periapsis and time limit for aerobraking until a suitable value was found within certain constraints of common sense and requirements of RFP.

During the aerobraking phase, after the orbit capture propulsion burn, small apoapsis burn maneuvers allow the vehicle to encounter the atmosphere at periapsis, lowering the energy of the orbit by a slight amount. The vehicle will now have a lower semi-major axis. These maneuvers continue until a semi-major axis of 19,390km is reached. At this point bus separation will occur.

separation

Structural attachment seraration is accomplished with double-ended tandem separation bolts. NASA Standard Initiator (NSI) pressure cartriges installed in each end of the bolt provides the explosive force to sever the bolts at a predetermined fracture plane. Energy absorbers are provided to absorb the energy of the separating halves. The bolts are designed to sever without debris, which might cause damage to components. Electrical disconnection is accomplished using the Range Safty System electrical cables with pullaway connecters, designed not to cause excessive torque[3.3]. These are current technology, ready to be used.

The separation sequence is initiated by a separation command signal, after a final command from earth, which energizes explosive bolts and allows a set of compressed springs to separate the bus from the lander. The spring device will not cause damage to delicate hardware the way thrusters might. The attitude control system compensates for disturbances and continues to aerobrake (see time table). At an altitude of 3677km, when the orbiter's scientific instruments indicate a healthy environment and approval from mission control, the entry software is initiated by the on board computer. A deorbit propulsive maneuver is executed and the vehicle begins to descend. Attitude control thrusters (8) are used to hold the lander in the proper attitude so that the aeroshield will protect the payload and instruments from the intense heat of entry and give a small amount of lift in the atmosphere(see AACS).

entry and landing system

The entry corridor at Mars, the window outside of which the spacecraft would either skip off the atmosphere if the flight angle is too low or will crash if the angle is too great, is large enough not to pose a serious guidance problem. Mars corridor width is from 850km to 2000km, depending on altitude. In reference, the earth's corridor is between 170km and 400km. Guidance tolerences imposed by different parabolic entry corridors can vary widely. Figure 3.4 shows the permissible deviation in flight path +-delta a verses altitude for different planets[3.4].

The lander encounters the sensible atmosphere at about 200km : peak velocity is about 15000 m/s. The lander decellerates increasingly and reaches a maximum decelleration at about 10km. At an altitude of 6400 meters, the lander has decreased to about 375 m/s. The parachute can now be opened. The parachute is a Dacron fabric, 54 meter parachute with 48 suspension lines. The size was determined using equation 3.2.

Equation 3.2 [3.5] $$Z = (2MG_0/pC_DA)^{1/2}$$

Z = Terminal Velocity

D = Diameter

M = Mass

C_D = Drag Coefficient

G_0 = Gravitational Constant

p = Density

It is deployed using a mortar device which accellerates a cord from the chute and opens the parachute. After the parachute is open and the vehicle is stable, the aeroshield is separated by an explosive bolt compressed spring device similiar to that used for the satellite separation. The main engine is also released at this time using the same method of separation. In about a minute, the lander drifts down to about 1200 meters and has a velocity of 60 m/s (eq. 3.2). The 3 terminal-phase engines are turned on and throttled to achieve the proper rate of descent (see Propulsion for specifications). The parachute is released and allowed to drift away. A radar altimeter and a radar unit determine the altitude and velocity respectively. This data is fed into the computer and the level of thrust required is output by the engines(eq.3.3).

Equation 3.3[3.6] $dV/dt = pV^2 G_0 / 2P_x - G(h)\sin Q - P\cos K$

V = Velocity

Q = Inclination Angle

h = Flight Altitude

P = Thrust

K = Flight Angle

This equation represents the equation of motion of the lander. The thrust is varied in such a way that the final velocity is a minimum, about 1.5 m/s. Two air bags are inflated with compressed air before touchdown (fig 3.6). Upon landing, at a specific internal pressure, pressure valves in the outer bag release air allowing for a soft landing. The inner bag provides support for the lander and protect it from possible damage from rocks. This configuration should be lighter and more stable than the conventional crushable landing legs, given the long payload to work with (fig. 3.6).

Upon landing, the lander will notify the Mars crew so that they may begin to remove the airplane from the container. The container will have a latch on one end to allow the end caps to be removed easily. Once the airplane is removed, the lander has reached the end of its useful life.

test concerns

The ellipsoidally blunted, raked off elliptically wide angle cone has several desirable factors,but it should be compared with other lifting brake shapes such as a blunted symetrical cone with an off-set center of gravity. Wind tunnel and computational fluid dynamic analysis will help assess the aerodynamic and aerothermodynamics of each configuration. Improved numerical flow field computations will improve shapes and designs, since wind tunnels are limited in their simulation capabilities at these high velocities and altitudes. This may evolve a more aerodynamicly and thermodynamicly stable shield.

Developments in composite structures would reduce the mass of the spacecraft without sacrificing strength or rigidity. Also, the use of lighter soft insulation having improved temperature and strength qualities would also reduce mass and allow for less thermal protection for components.

The air bag concept for soft landing needs testing to determine its feasibility. This area is a new concept with very little backround in an aerospace environment. Strength tests need to be run as well as puncture and thermal resistance tests.

subsystem interactions

Communications with Propulsion, AACS, Structures and Mission Planning were an intergal part of this design. Mission Planning gave me parameters for AEROB, decided the orbit and determined the separation timeline. Interaction with Structures accomplished the design and integration of the components, as well as deciding on the materials of the components for aerobraking and stuctures. Propulsion determined the size of fuel tanks and batteries needed for power during the descent phase for the mission, after being given power requirements(fig 3.5). Propulsion also picked descent rockets for the terminal phase after being given thrust requirements. AACS integrated the control portion of the final descent, once given attitude requirements for the entry.

requirements

Table .1

	orbit capture	entry	separation	off the shelf	before 1998	greater than 4 yr.	art. int.	reliable	low cost min. mass
aerobrake	x	x			x	x		x	x
pyrotechnics			x	x	x	x		x	x
electric disconnects			x	x	x	x		x	x
parachute		x		x	x	x		x	x
structure	x	x	x		x	x		x	x
guidance aacs	x	x			x	x	x	x	x
thermal protection	x	x		x	x	x		x	x
thrusters	x	x		x	x	x		x	x
spring			x		x	x		x	x
radar		x	x	x	x	x		x	x
altimeter		x	x	x	x	x		x	x
landing engines / bag					x	x			x
antenna		x		x	x	x		x	x

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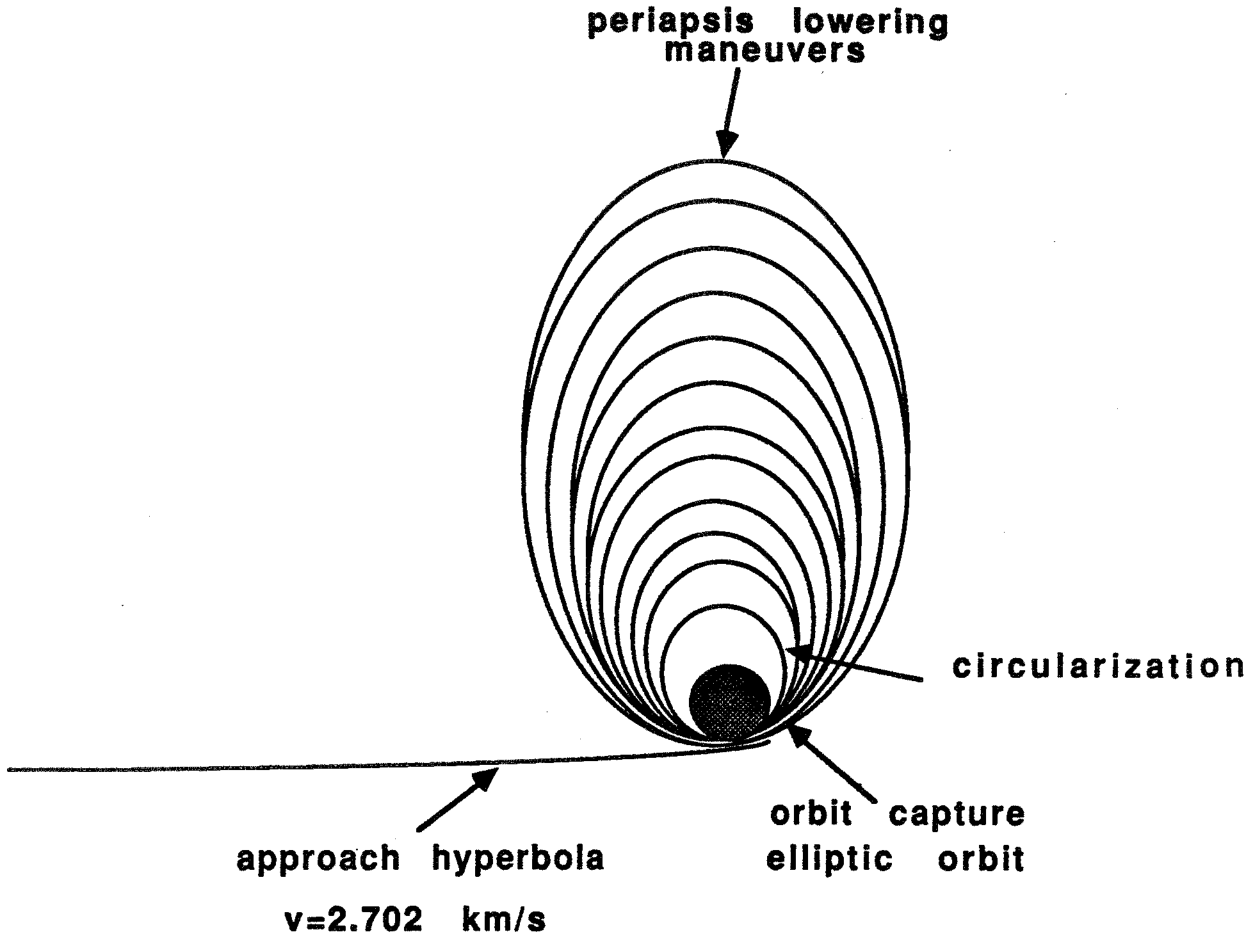
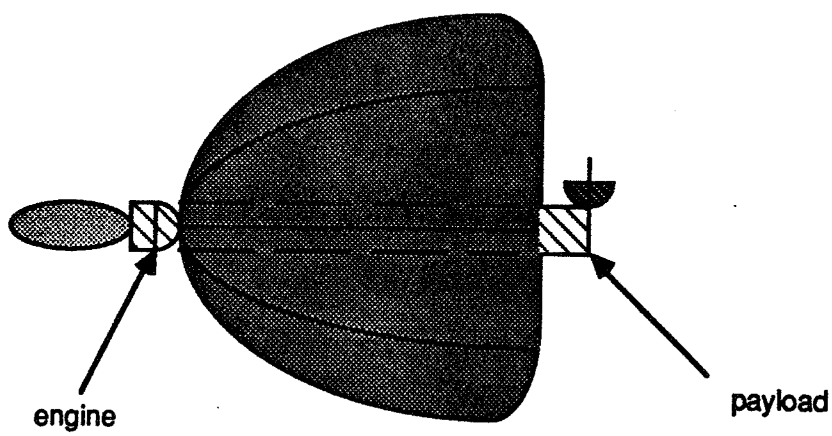


figure 3.1



ballute

figure 3.2

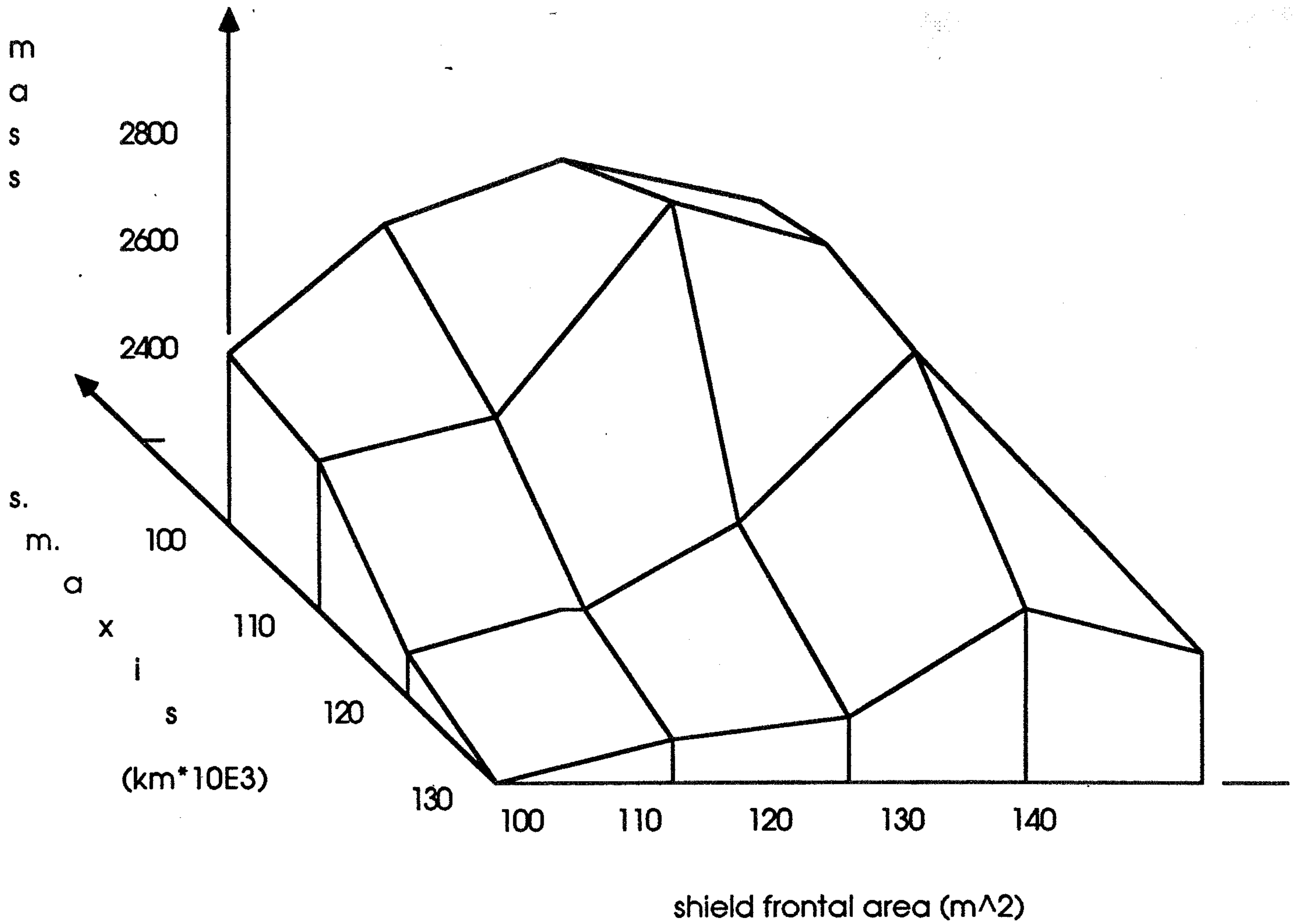


Figure 3.3

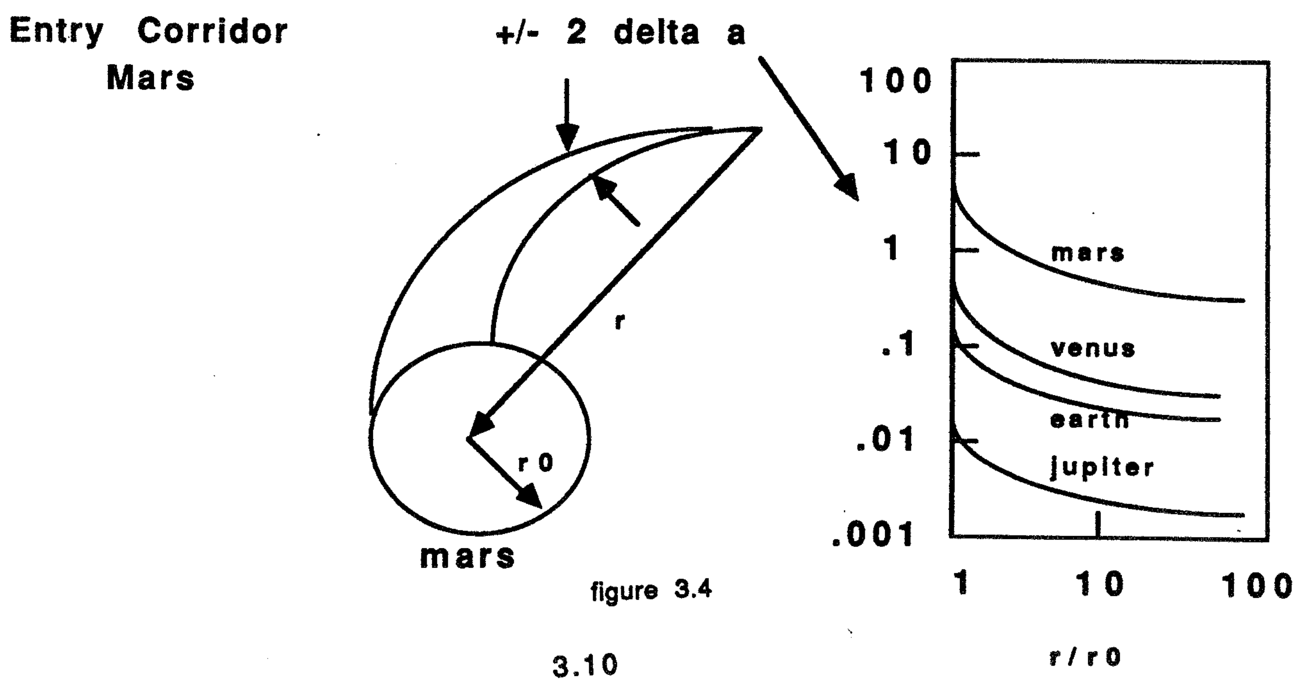


figure 3.4

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Component	mass (kg)	power req. (watts)	description
aeroshell	474	0.0	14m x 11m ellipse
structure	78	0.0	stiffners/braces
power suppply	2.2	0.0	batteries (2)
data and control	6.0	10.0	computer
aacs	30.0	52.0	thrusters/gyro
pyrotechnics	3.5	2.0	separation bolts
cabling	30.0	0.0	electrical/fluid
propulsion	296.2	8.0	fuel-104.2 kg engines -3 @ 64 kg
temperature control	5.9	0.0	insulation
antenna	3.0	8.0	UHF C3 link
parachute/mortar	85.0	2.0	54m chute 48 susp.lines
terminal descent	75.0	10.0	engines /air bags
radar/radar Altimeter	24.0	70.0 + 29.4	Ranges 2-20,000m / 3-20,000,000m
TOTALS	1096.8	191.4	

figure 3.5

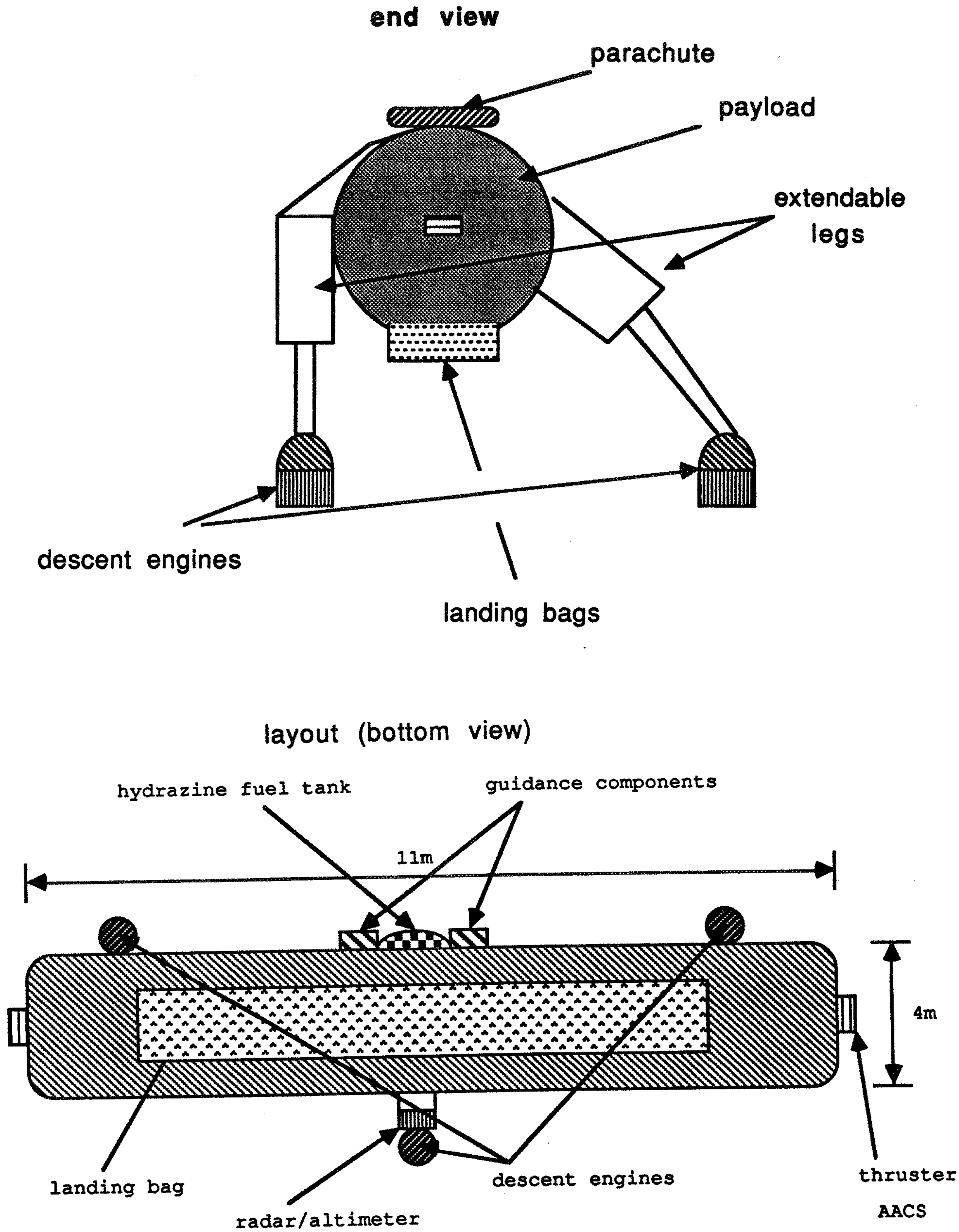

figure 3.6

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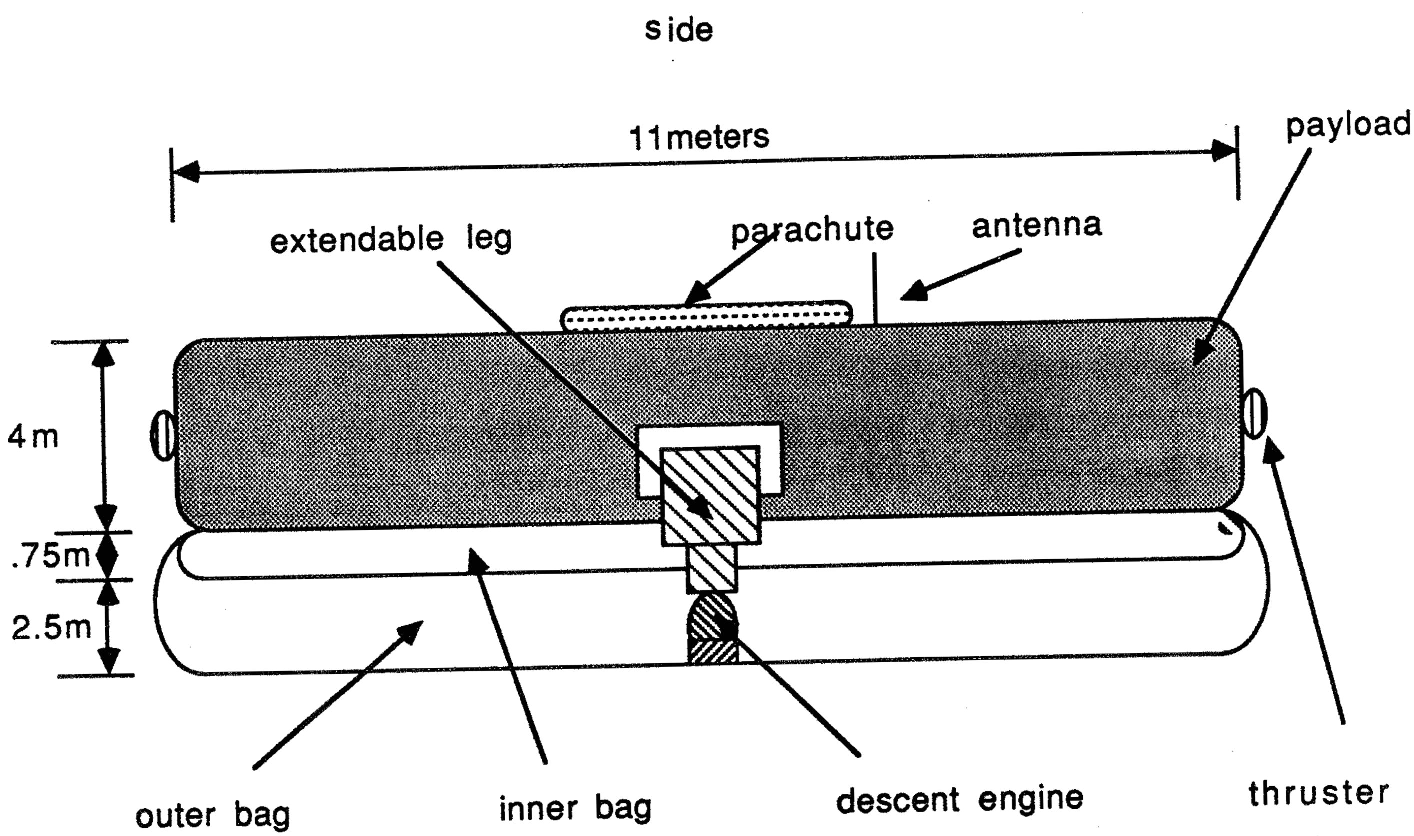

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3.3 Separation Subsystem AAE241

3.4 Chapman, Dean R.

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3.6 Ivanov, N.M., Martynov, A.I. and Belykh, V.D. "The Effectiveness of Using a Minimum-Final-Velocity Criterion in the Synthesis of Spacecraft Descent Trajectories Onto the Surface of Mars." AIAA Paper No.73-889.

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IV. STRUCTURES

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The structures subsystem provides basic physical support for all the components of the spacecraft. The actual layout of the spacecraft is determined by the capabilities and limitations of the various instruments and subsystems involved.

Examining the power requirements first, the solar arrays are very large. They must be placed away from the spacecraft to maximize the area exposed to the sun and to avoid blocking the fields of view of the science instruments. Another aspect of this subsystem is propulsion. The tanks containing the propellants must be located so as to minimize the distance the propellants must travel. The thrusters must also be placed in positions to both minimize the travel distance of the propellant and to provide effective control of the spacecraft. For this reason they are placed as shown in Fig.4 .1.

With respect to the aerobrake, a material must be found that is able to withstand repeated applications of extremely high temperatures. In order to withstand the extreme temperatures of the multiple orbit aerobraking, the material must provide high strength and stability at high temperatures. The standard material to satisfy these types of requirements is a carbon-carbon composite material which has been used in such applications as the Space Shuttle heat shield.

In order to satisfy both the AACS and communication requirements, the instruments chosen in each of these subsystems must have clear fields of view. The AACS star trackers and mapper and the sun sensor are placed so as to be able to locate their respective targets, and the high- and low-gain antennas are placed on the periphery of the spacecraft to avoid disruptions of communication caused by spacecraft instruments. Guidance computers and gyros may be placed wherever is necessary to optimize the inertia matrix. Also, to fulfill science requirements, the magnetometer must be placed away from the body of the spacecraft to avoid extraneous electromagnetic radiation. Other science instruments must be placed so as to have a clear view of the Martian surface and atmosphere.

SYSTEM INERTIA

The entire delivery system was split into two major components for simplification of component placement about an acceptable center of mass. The first component identified was the instrument bus. The inertia tensors of the various instruments chosen for the satellite were entered into a program designed to optimize component placement. The results of the program yielded the following satellite inertia tensor:

Final Design

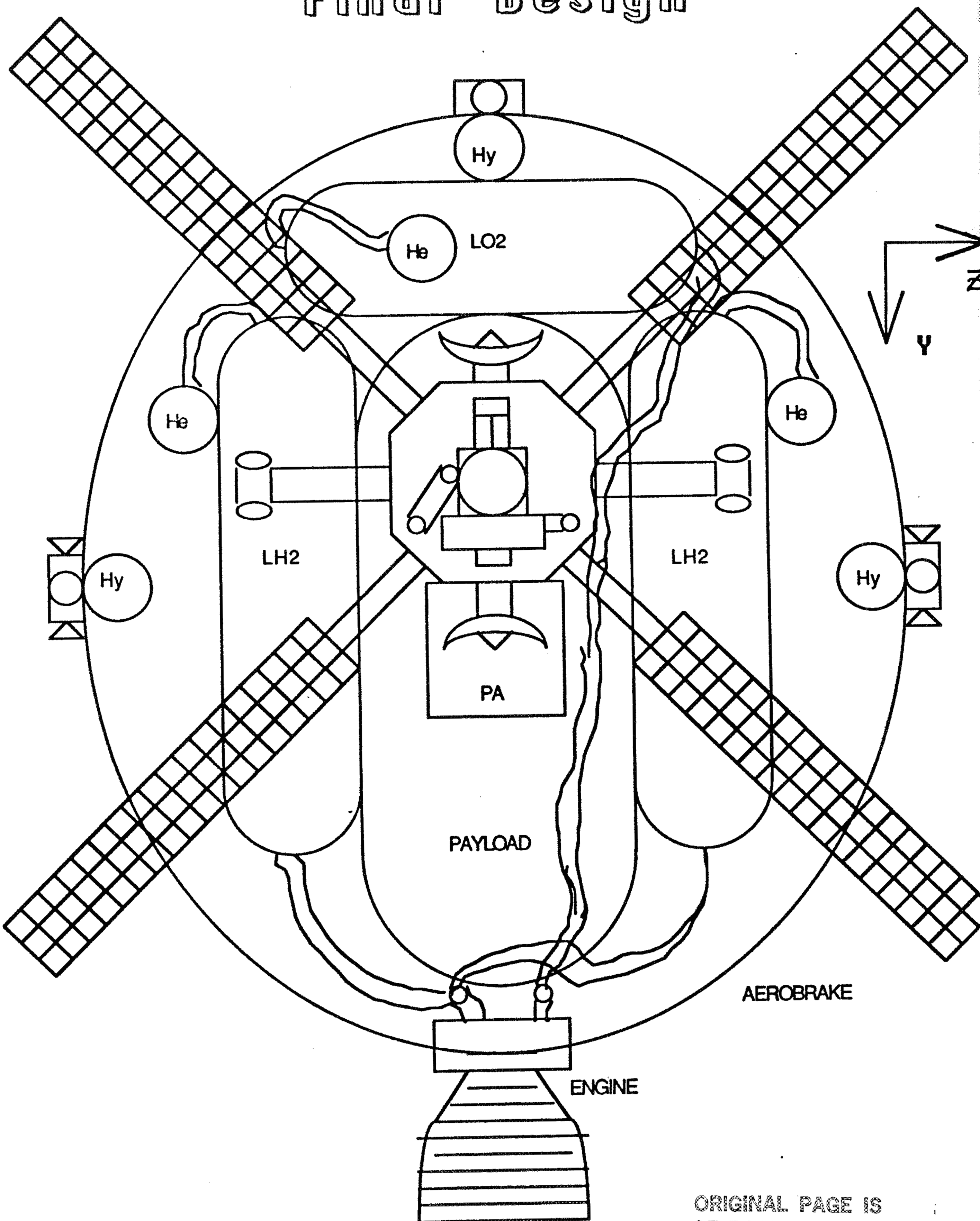

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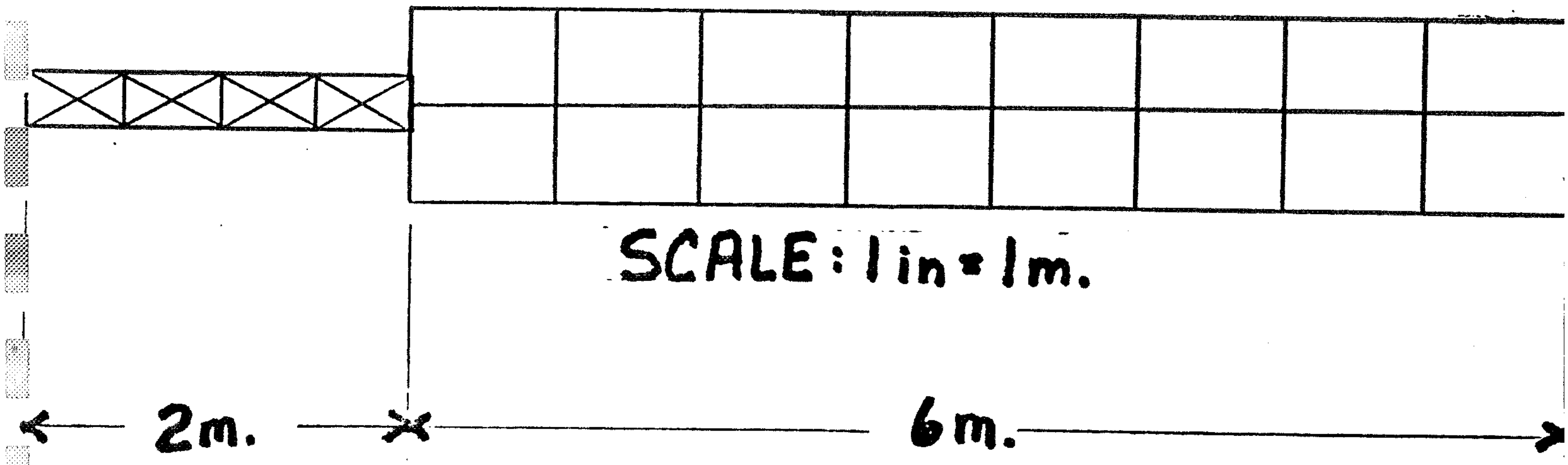
FIG 4.1

DEPLOYED:

EACH SOLAR ARRAY MASS = 8.1 Kg

EACH SOLAR ARRAY BOOM MASS = 6.3

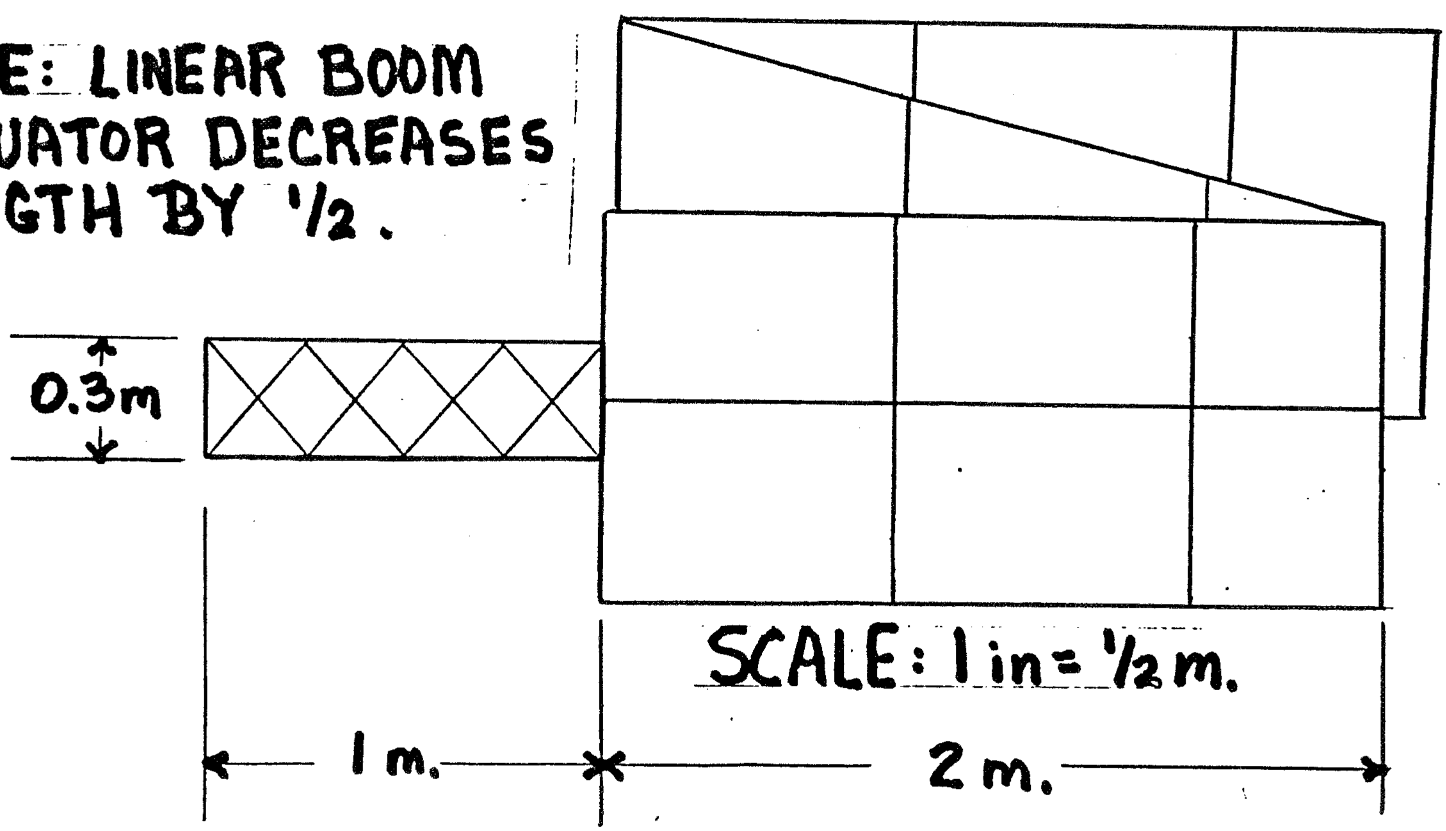


NOTE: FOLDING POINTS

(SOLAR ARRAY FOLDS UP INTO 2m. x 1m.)

RETRACTED:

NOTE: LINEAR BOOM ACTUATOR DECREASES LENGTH BY 1/2.



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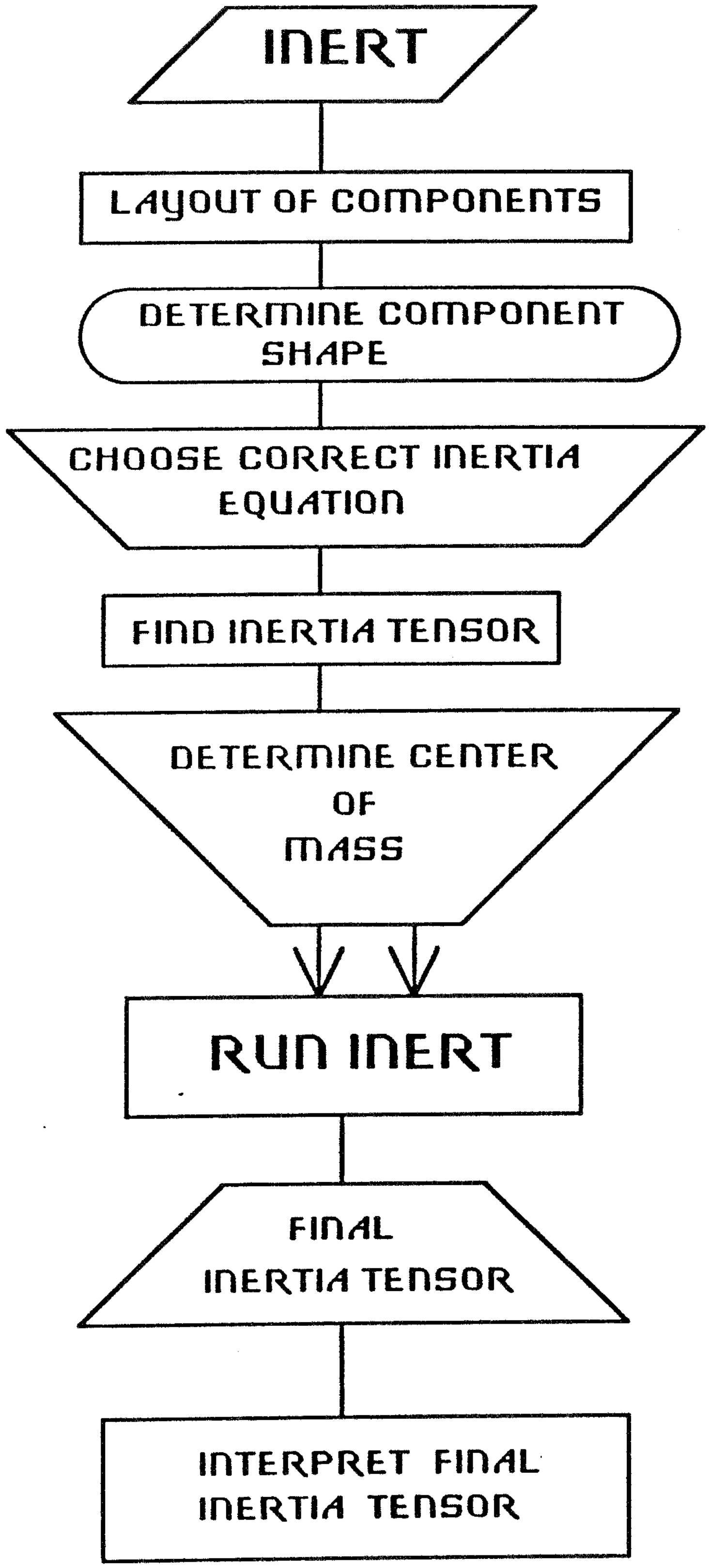

INERT
LAYOUT OF COMPONENTS
DETERMINE COMPONENT SHAPE
CHOOSE CORRECT INERTIA EQUATION
FIND INERTIA TENSOR
DETERMINE CENTER OF MASS
RUN INERT
FINAL INERTIA TENSOR
INTERPRET FINAL INERTIA TENSOR

M.O.A.

4.4

12993.04	0.3989	0.0667	
0.3989	4578.60	-0.9825	kg-m^2
0.0667	-0.9825	6348.40	

This, in turn, yielded the satellite system center of mass, found to be:

X = -0.0034 m, Y = -0.0529 m, Z = 0.0079 m,

Mass = 1077.21 kg

Therefore, when the satellite is deployed it is very nearly centered about its geometric center of mass.

The second component defined the remaining elements of the delivery system, including the aerobrake, payload, and fuel tanks. A similar procedure was performed and yielded the following spacecraft inertia tensor:

561439.04	-46492.24	183.72	
-46792.24	104965.70	1247.49	kg-m^2
183.72	1247.49	498880.53	

Corresponding to a system center of mass of:

X = -1.7039 m, Y = -3.3036 m, Z = -0.0083 m,

and a total mass equal to approximately 24,000 kg.

In closing, the placement of the components of the entire system yields approximately a stable configuration.

THERMAL CONTROL

There are two fundamental methods for controlling spacecraft temperatures: passive thermal control and active thermal control. Passive thermal control of a system and its components is obtained wholly through geometrical design and the selection of materials with the required thermophysical properties. This method also includes the static use of temperature-induced physical changes in materials. Neither power or moving parts are employed.

Passive thermal control is more reliable than active thermal control. No moving parts or switches are used, thus eliminating malfunctions. The passive system usually involves less mass as well. The passive control system is usually identified by what is known as Curie point transition. The Curie point transition utilizes first order atomic transition which takes place as a function of temperature. The obvious major advantages to this system are reliability, simplicity and small weight factors.

The use of a thermal wick is also being considered to aid in the cooling of the spacecraft components (see Fig. 4.2). A heat transfer fluid contained within the housing of capillary channels is evaporated at the equipment which is cooled and condensed at the vehicle skin. The liquid is then returned to the hot end of the system by these capillary channels. Although simplistic in design, sometimes problems lie in placing the heat transfer fluids within the channels.

The system chosen is called solid cryogenic refrigeration (SCR). The system utilizes the heat of sublimation of selected materials to cool selected spacecraft components. Conceptually, these systems consist of a container of solid refrigerant which is thermally isolated from everything except the component which is being cooled. SCR is well suited to accurately control components at low temperatures. The component temperature is essentially fixed by the refrigerant temperature. Although a small possible problem lies in maintaining the ambient pressure, SCR is virtually problem-free.

MATERIALS SELECTED

In selecting the proper material suitable for both a space journey and a reentry situation the following requirements must be met:

Space cruise:

1. Suitable behavior of the material at low pressures.
2. Must be able to guard against radiation damage.
3. Must be able to resist impact of micrometerites.
4. Friction seals and bearing surfaces must not yield.
5. Need to keep in mind lightweight material for vehicle construction in order to comply with overall mission objectives.

Reentry phase: Need to be concerned with:

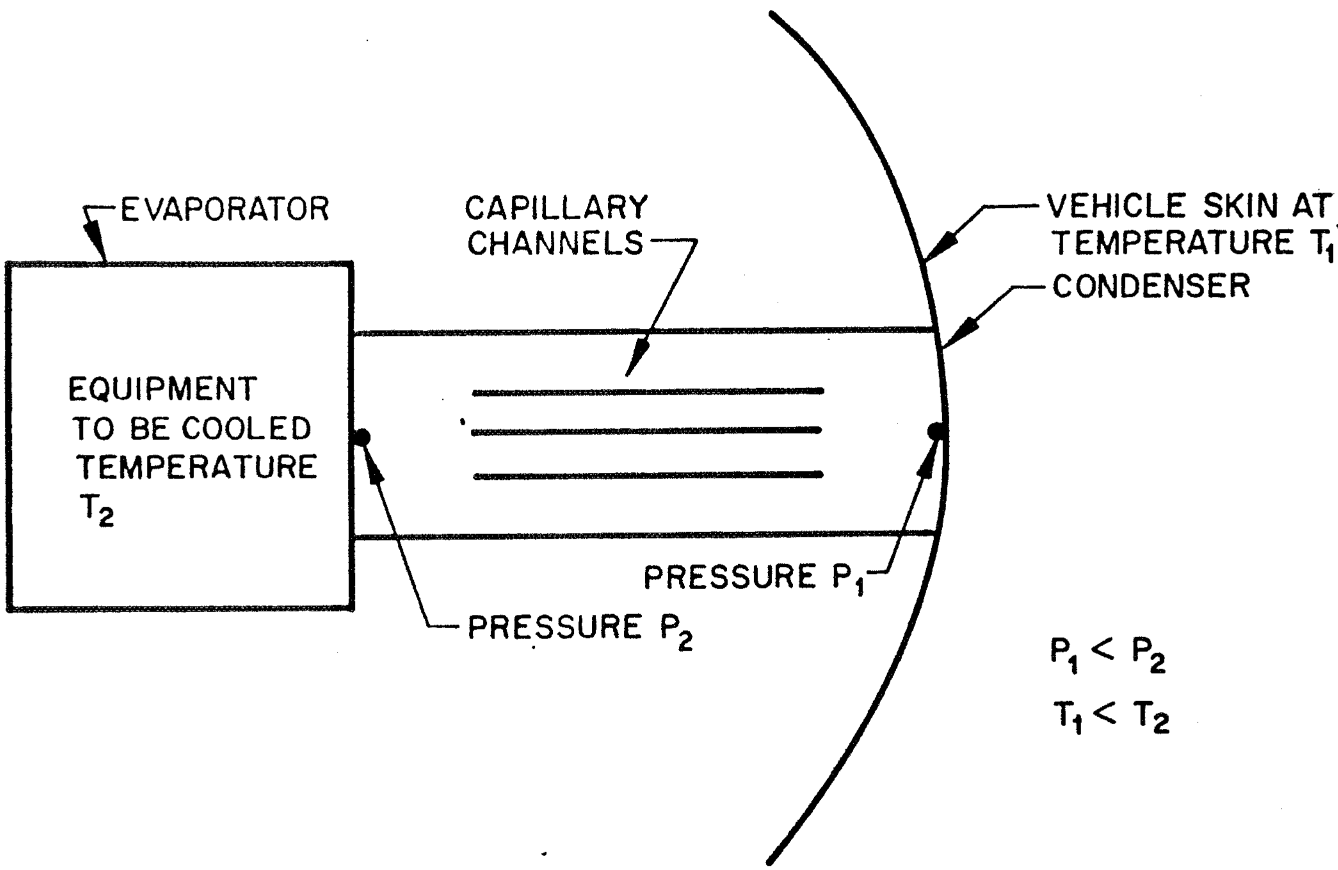
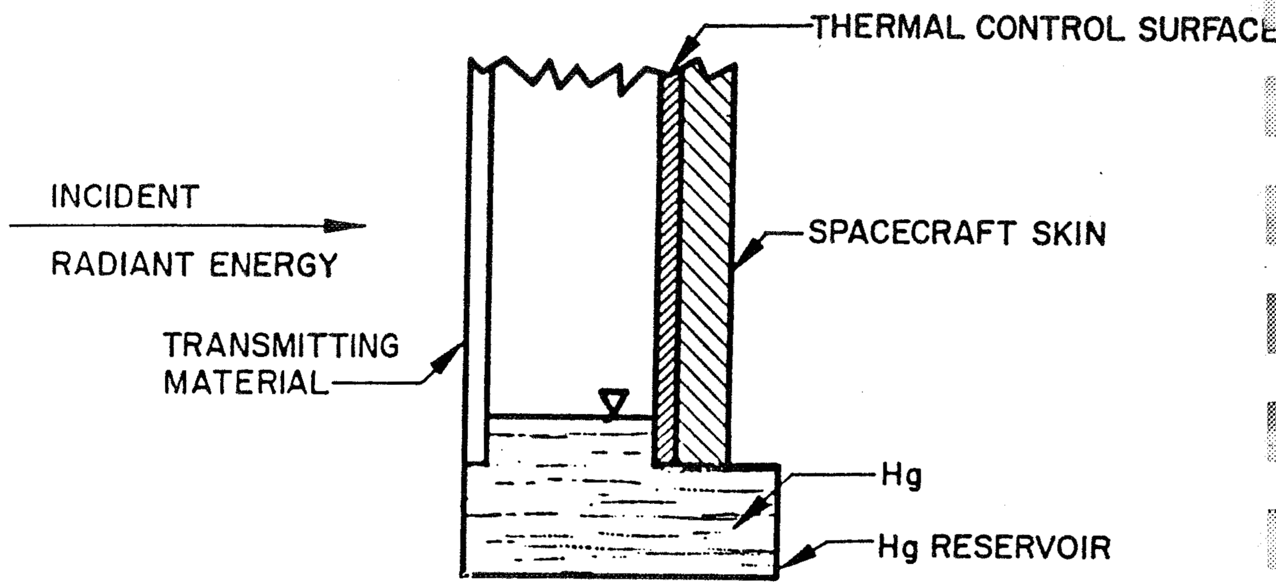
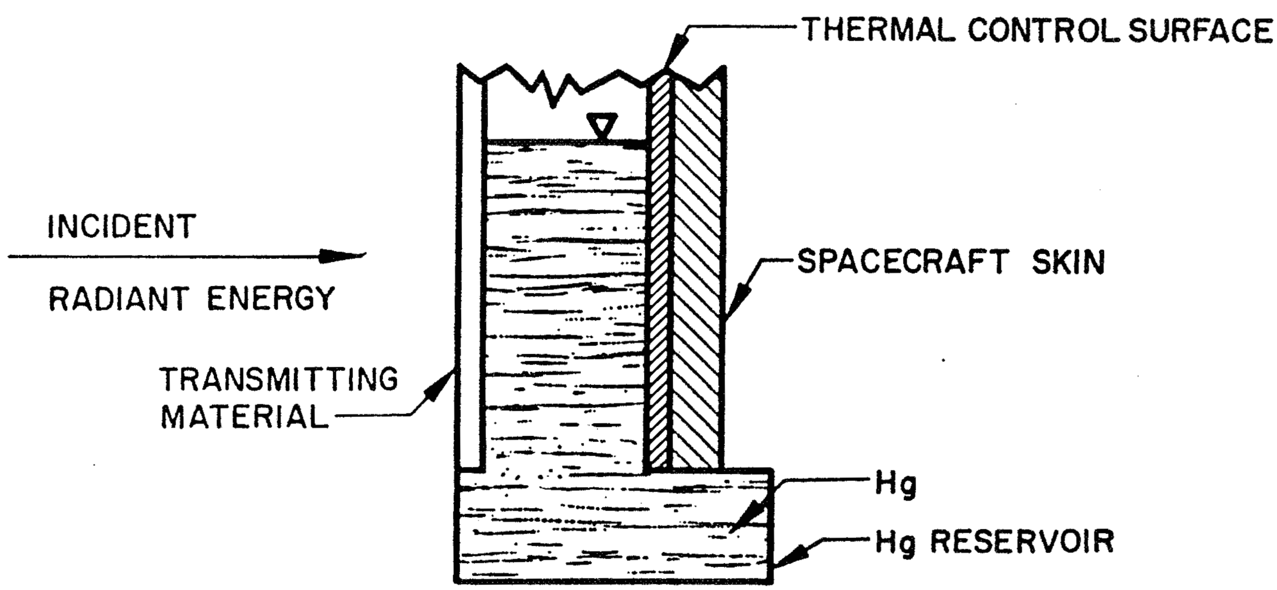


Fig. 4.2 Thermal Wick



(a) Hg SHUTTER IN COOL POSITION



(b) Hg SHUTTER IN WARM POSITION

[redacted] Mercury Shutter

1. Thermal stresses

2. Material survival under thermal shock.

3. Erosion, ablation, and heat sinks

4. Want to look at thermal insulation and protective coatings.

Several materials were considered to determine which could best suit the needs of this mission. These are tabulated in Table 4.1. Although Table 4.1 yields a boron-epoxy or a graphite-epoxy to best suit mission needs, they were not selected because they are still in the experimental stages of development and are not expected to be available before 1998 for particular immediate structural needs. The next best choice was the material titanium. When incorporated into a structural design a honeycomb-type filler configuration is used. The structural core material used is adhesively bonded between high strength faces producing a sandwich panel with the highest known strength to weight ratio of any known material. Titanium is strong, lightweight, shapable, corrosion and fatigue resistant, and versatile.

SHUTTLE COMPATIBILITY

For adequate shuttle compatibility, unassembled components to be delivered to the on-orbit space station repair and assembly platform must be fitted with trunions. To insure safe delivery, that is, undamaged parts on any of the components, the trunions must be placed where they can handle the greatest amount of force disturbance associated with the extreme stresses involved during shuttle launch and cruise. At the same time they must be placed where they will be directly compatible with the fasteners located at various horizontal distances along the shuttle cargo bay. The location of the trunions are indicated in the various component diagrams in this section of the paper and are placed to meet the above constraints.

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TABLE 4.1

LIST	SPECIFIC STRENGTH	DENSITY	MODULUS OF ELASTICITY	TOTAL
ALUMINUM ALLOY 7000 SERIES	7	6	12	25
BORON-EPOXY, 55 V/O BORON	2	4	7	13
CARBON STEEL	11	8	8	27
COLBALT AND ITS ALLOYS	8	12	6	26
~~GRAPHITE-EPOXY, 56-60 V/O GRAPHITE~~	~~1~~	~~1~~	~~4~~	~~6~~
MAGNESIUM ALLOYS	10	2	13	25
NICKEL AND ITS ALLOYS	9	11	5	25
STAINLESS STEEL	5	10	8	23
POLYESTERS, THERMOSET PULTRUSIONS	3	3	14	20
TITANIUM AND ITS ALLOYS	4	7	10	21
ULTRA-HIGH STRENGTH STEELS	6	9	11	26
BERYLLIUM AND ITS ALLOYS	12	5	3	20
MOLYBDEDIUM AND ITS ALLOYS	13	13	2	28
TUNGSTEN	14	14	1	29

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V. ATTITUDE AND ARTICULATION CONTROL

The attitude stabilization of either an orbiting satellite or an interplanetary transporter system is one of the most critical aspects of their designs. The increased diversities of the missions as well as the ever-shrinking tolerances on the attitude accuracy requirements make this extensive engineering task even more of a problem.

The problem of design begins with a list of system requirements, each of which must be satisfied before a design is complete. The fulfilling of the requirements must be based on engineering studies. Rather than picking randomly from a catalog of components, selection must be based on trade studies. Trade studies are the weighing of advantages and disadvantages of each prospective component to arrive at a selection that will be the optimal choice for the specific aplication.

Requirements

The requirements that will guide the design of the attitude and articulation control system (AACS) are given by the RFP:

-design should use off-the-shelf hardware where available

-nothing in the design should preclude it from performing different missions or carrying vastly different payloads to different destinations.

-design should not use materials or techniques expected to be available after 1998.

-design should stress simplicity, reliability, minimum mass and low cost.

-design lifetime of four years, but nothing in the design should preclude it from exceeding this lifetime.

In this section it will be shown how each of these requirements were met as well as the technical approach involved in meeting them. On a somewhat more specific level, a second list of requirements was considered. This system should:

-send telemetry to C^3.

-accept and act on commands from C^3.

-receive power from the PPS.

-accurately control and point the vehicle.

-effectively minimize weight and cost.

-maximize reliability, accuracy and efficiency.

Analysis of Candidate Systems

Since the design of the NOMADS calls for a large mass (see Structures subsystem section for quantitative analysis), it was decided that two separate AACS designs should be implemented. One design is for the control of the entire vehicle en route to Mars. The second system is the attitude control for the instrument bus that will be placed in an orbit about Mars.

The AACS of most vehicles can be classified into a few main groups of like systems. The first and indeed the most important step in a design is to pick from these groups a system that will best be suited for the specific application.In the following pages the choices for each AACS component will be analyzed.

A system of solar vanes was examined first. This system relies on solar radiation pressure for the attitude control of a vehicle. For applications concerning the transfer vehicle the solar vanes would be very impractical since these systems are normally implemented solely for station keeping maneuvers. The large mass and high velocity of the vehicle excludes this system for a design choice. This mission also requires that the instrument bus platform make many scientific observations from orbit. The instruments that will collect the data are mounted facing the Martian surface and actuation of these instruments produces disturbances in the system on a regular basis. The many subsequent control demands that will be placed on the AACS due to the disturbances of the instruments' actuations far surpasses this system's ability to maintain an acceptable level of dynamic control integrity ,due to the fact that it is considered a passive system. That is to say, a system that does not rely on an energy source, sensors, and/or artificial feedback for attitude reference or control. For these reasons solar vanes were excluded from the design choices.

Spin and dual-spin systems have successfully been flown on past missions but presented several serious problems for our design. As previously mentioned, the scan platform must remain very still to allow the collection of accurate data by the science instruments. The stringent requirements on the hold or pointing mode make this system unusable immediately (see Science subsystem section for accuracy requirements). In addition to the accuracy, the power requirements are such that the solar array configuration would need to be over 5 meters long and given the radius of our bus (approximately 1.5 meters) this would mean that the satellite would be spin stabilized about the axis of least moment of inertia. With this in mind, it is easy to see that there is a good chance that a disturbance,such as fuel slosh, instrument actuation, etc., could cause the satellite to tumble. This system was rejected as an AACS candidate.

The design has been narrowed down to a choice between two systems, namely: three-axis active stabilization and reaction wheel stabilization.

NOMADS is a medium-size satellite. The three axis active would provide a highly accurate control; however, there are some penalties. For a misson of four years the fuel required for station keeping would be quite large, and considering that the science instruments will be actuated relatively often, the fuel requirements would increase drastically. The equation used to estimate this systems mass is:

$$M_s = 65 + 0.62\,(M_{s/c} - 700). \tag{5.1}$$

REF(8)

From this analysis we find that the mass of the required system would be approximately 65kg. Shortly it will be shown that the chosen system of reaction wheels has a mass of 61.0 kg. Immediately an advantage in mass is realized through using the reaction wheel system .This advantage becomes much larger when the extra fuel required for the three-axis stabilized is taken into consideration.

Reliability is an extremely important issue in this design because of the four year mission length. The three axis active system is the more complex of the two candidate systems and because of this fact it is the least reliable of the two. These trades and more,shown in Fig. 5.1,indicated that the reaction wheel system was the appropriate system for this mission.

Satellite System Component Selection

Now that a system of reaction wheels has been chosen as the AACS on the NOMADS, the subject of wheel unloading must be addressed. Two systems for unloading were considered: boom unloading and thruster unloading. The characteristics of each are shown in Fig. 5.2. In addition to the disadvantages in Fig. 5.2, the boom unloading posed another problem. The design was becoming much too crowded with four solar arrays, two instrument booms and two antennas. Since the boom would have had no place to be mounted effectively, boom unloading was ruled out.

Turning to the system requirements, we first are concerned with the mass of fuel that is needed. The instrument bus will, upon reaching its designated orbit, be required to detach and move away from the vehicle. To do this we must accelerate away from the vehicle and then decelerate to an on station hold position. For this maneuver an acceleration profile shown in Fig. 5.3 was chosen. Then using the equation for that profile and integrating, we get $\Delta V = 15$m/s. The fuel consumption for this maneuver is found by:

$$\Delta V = g_o I \; \ln(M_i/M_f) \tag{5.2}$$

REF(11)

with $\Delta V = 15$ m/s, $g = 9.8$ m/s, $I = 225$, and $M = 600$kg

In eqaution 5.2 I= the specific impulse for the monopropellant Hydrazine. The reason Hydrazine was chosen is because of its long history as a dependable propellant and the fact that it is a monopropellant requiring only one fuel tank and a small pressurant tank. Fuel mass requirements for similar delta V's were calculated, and the tank weights for each type of fuel,bipropellant and monopropellant, added to this. It was found that for the required delta-V of this mission Hydrazine was the optimal choice.

Tank sizing was performed by using the equation:

$$M_f = 4/3\, \pi\, r\hat{}3 \qquad (5.3)$$

REF(11)

A total mass of fuel estimate was obtained through the use of the equation:

$$\Delta m = (Iw)/(g\, I_{sp} l) \qquad (5.4)$$

REF(10)

where I = moment of inertia about a specific axis, w = angular velocity (which for the purpose of this analysis was chosen to be 2π rad/day),g = 9.8 m/s^2, I_{sp} = specific impulse of Hydrazine = 225 sec , and l = moment arm of thrusters = 1.5 m.This analysis yielded a total mas of M = 80 kg and tank size of r = 0.27 m. These estimates are based on a +20% error factor.

For a spacecraft, H = Iw , where H = angular momentum imparted to the body by a reaction wheel, I = the body's inertia matrix, and w = angular velocity, or slew rate. A slew rate of 0.3 rev/min was chosen which corresponds to an H of approximately 2.0 ft-lb sec. In choosing a reaction wheel it was desirable to minimize weight, size, and operating rpm's while still meeting the slew rate requirements. The rpm's were minimized in order to give added reliability and life expectancy to the components (refer to Component Table 5.1 for the chosen component parameters).

Thrusters were chosen on the basis of weight, operational life, total impulse, number of pulses, and past performance. NASA's standard 1 lb thruster was chosen on the basis of the above qualifications.

An inertial attitude reference system is essential to the mission. For this component low weight, low power requirements, low noise, long life and accuracy are needed. Research revealed a new gyro based on fiber optics, the Fiber Optic Rotation Sensor (FORS). This component optimized all parameters and has been chosen for this mission (refer to Component Table 5.1).

An onboard computer was a necessity for the complex attitude controls in this mission. The computer will be selected only from units that are compatible with the C3 components with wich it must interact . Optimization of weight, operational life, reliability and memory led to the selection shown in Table 5.1.

Accurate sensing is a very important aspect of maintaining an operational satellite. It was decided that a fine sun sensor in conjunction with a coarse sun sensor for each side of the satellite should be used. This redundancy is a precaution taken to ensure that in the event the satellite stabilizes after a tumble it will always be able to orient itself to the sun. The coarse sun sensor must minimize size and weight while having relatively good accuracy. The coarse sun sensor should also have a 180 degree field of vision to ensure complete scanning capabilities. The fine sun sensor must have a narrow field of view and a very good level of accuracy. Refer to Table 5.1 for selected component.

For attitude and position reference a star tracker was chosen. This unit was chosen in much the same way as the other components. Here again, weight, size and power requirements should be minimized and pointing capabaility maximized. This instrument also requires a gimballing mechanism which should also optimize applicable parameters.

Almost complete, the satellite design needs a number of actuators for the instrument booms and antenna. An accuracy optimization was the most important study in this selection process; however, weight, size and powerrequirements were also looked at. Two linear boom actuators were chosen as well as an antenna actuator. Refer to Table 5.1 for the antenna actuator parameters and to the Command and Data Control subsystem section for pointing accuracy requirements.

Transfer Vehicle Component Selection

As was discussed in the introduction to this section, two independent AACS will be used. For the transfer vehicle a three axis-active system was chosen. The momentum wheels could not provide the necessary delta-V trims needed and a properly sized thruster system for wheel unloading would not be effective either. The thrusters will be bipropellant units to make it possible to draw from the main propellant tanks as a fuel supply. The additional fuel requirements were calculated using Eqn.5.4 as before. Here, it is assumed that the craft must be able to complete a 360 degree maneuver every 24 hours. Therefore $w = 2\pi$ rad/ day. The delta-V trims needed were estimated from past missions. Fuel for reentry was estimated in this manner as well. A 20% error facter was then added.

The sensors chosen were the same as used on the bus with a slight loosening of the accuracy tolerances. Only two sun sensors were called for . A star mapper was added to this vehicle for additional data to pinpoint the vehicle's position. The gyro,computer, and star tracker are similar to the units on the bus and were selected through the same process.The thruster chosen is a NASA standard 5 lb thruster. This system will produce a slew rate of approximately 2.5 rev/min on the NOMADS.

Attitude Modes

The AACS design now complete, the modes of operation for both systems will be defined and examined. The attitude modes common to both the transfer vehicle and the satellite are: delta-V trims, fine point / hold, slew mode / maneuvers, star acquisition and sun acquisition . During the delta-V trims,the thruster system will fine tune the final positioning of both vehicles. For the transfer vehicle , delta-V trim mode is very important due to the fact that the main propulsion system of the of the craft is not capable of precisely controlling the positioning of the vehicle. Although the thrusters on the satellite are used for the main propulsion as well as delta-V trim, mode differentiation is seen by the onboard computer in that the accuracy of control is greater for the delta-V trim.

Fine pointing or hold is a mode in which extreme accuracy is required from each component in the system. In this mode gyro angular velocity is usually increased to induce greater stability in the component at the expense of an increase in power consumption. It is in this mode that three-axis active stabilized is the most accurate, while technological advances in reaction wheel systems will soon bring momentum stabilized systems up to a very respectable accuracy level- such as seen in this satellite design.

The slew or maneuvering mode is similar to the fine pointing in that the gyros are sped up for the maneuver. In this mode, calculations and corrections for fuel slosh and structural bending and flexing must be done either at the design level or in the immediate mode of the onboard computer. In this design the problem is left to the computer.

Star acquisition and sun acquisition are similar modes of the AACS. During these operations the attitude system will be fed data from the star tracker and sun sensor respective to each mode. This information will be processed by the computer, the computer in turn will send commands to the rest of the system to actuate the pointing of the instrument and possible reorientation of the vehicle. This mode is a kind of hybrid in that after the data is sent to the computer, the vehicle goes into a mode that is essentially identical to the fine pointing mode.

There will be a cruise mode for the transfer vehicle. In this mode the AACS will perform a constant monitoring of attitude and the main propulsion system will move the vehicle.

In this mode, the satellite will be mostly dormant with only a very few components in operation. The sun sensors on one side of the satellite, the upward side, will be used for attitude sensing and solar array pointing. The star tracker will also be in use furnishing attitude and position data. Actuation of the communications antenna is necessary throughout all modes of this mission.

The satellite will have a mode that is unneccessary on the transfer vehicle, that is momentum wheel unloading. In this mode a torque will be applied to the momentum wheels to slow it down while the thruster system creates a torque in the opposite direction. Through this coupling of torques it is hoped that the satellite will not experience excessive motion about any axis.

Component Placement

Placement of sun sensors, star trackers and the star mapper will be selected on the basis of each individual instruments' field of vision, interaction or interference with other components and weight distribution requirements. The thrusters on both the satellite and the transfer vehicle are placed such that they will generate the greatest torques on each body.Since torque is given by :

$$T=Fl \qquad \text{FIG(5.5)}$$

maximizing torque is done by increasing l, the moment arm. Thruster force is denoted by F. The thrusters were located axi-symmetrically so as not to introduce rotations along any other axis other than the intended axis.Thrusters were also placed symmetrically about the center of gravity so that their masses would not generate any non-axial terms in the vehicle's inertia matrix. Each thruster on board the transfer vehicle was mounted on a small boom assembly. This assembly extended the thrusters out over the edge of the heat shield for attitude control during the flight to Mars. During the reentry of the vehicle , the booms will retract the thrusters to a position just behind the outer lip of the heat shield,safe from the extreme heat generated during reentry. In the retracted position the thrusters pointed in the downward direction will not be used. Thruster placement is shown in Fig.5.5.

The reaction wheels must be placed directly about the center of mass of the satellite for most accurate control and maneuvering. Gyro and computer placement should be inside the body of the satellite,protected from the extreme conditions of the invironment. These two components will be placed at the discretion of the Structures subsystem analyst as well.

The control requirements placed on the AACS by the scientific instruments were compared to the accuracy of the designed system and it was found that the 2urad overall system pointing capability would be quite sufficient. The antenna pointing accuracy requirements (tolerances can be found in the Command and Data Control subsystem section) were met by both the overall system pointing capability and the antenna actuator capability.

AACS Interaction

The AACS must interact with all the other subsystems in the design. The structures system required component placement interaction. Each component had to be accounted for in the mass and inertia analysis. Also, some of the components of the AACS had to be mounted using special brackets.

The science subsystem required pointing accuracies that had to be met by the AACS. Aerobraking into orbit and reentry to the surface required attitude control consideration. Also the power and propulsion subsystem was given power and fuel requirements that had to be met before the AACS could function properly.

All of the interactions between the AACS and the other subsystems were monitored by the C^3 system. Commands from Earth ground control were received by C^3, then relayed to the AACS. Information as to actuation, attitude sensing and position were given to C^3 by the AACS, which then sent the data back to Earth.

Technical Problem Areas

Each component of the AACS meets requirements given in the RFP as well as the particular subsystem requirements as given at the beginning of this section. There are several tecnical problems facing the AACS design. The effects of the Martian atmosphere on the components should be analyzed. Also the effects of structural flexibility should be addressed; however, both of these studies are beyond the scope of this paper.

An effort to more accurately estimate the delta-V's associated with station keeping, manuevering and delta-V trims should be made utilizing a technical approach. Along these same lines an accurate estimate of momentum wheel unloading frequencies should be found. Once these problems are solved , this design should produce a very effective attitude control system.

ATTITUDE CONTROL SYSTEM TRADES

	SELECTED SYSTEM
3-AXIS ACTIVE PERFORMANCE ADVANTAGES	REACTION WHEEL PERFORMANCE ADVANTAGES
-WELL SUITED FOR MANEUVERING -HIGHLY ACCURATE POINTING -FAST CONTROL ACTUATION -EFFECTIVE ON SATELLITES WITH LARGE MASS AND INERTIAS -CAN AID IN DELTA-V TRIMS -LESS MASS FOR REDUNDANCY UNITS	-COMPONENTS LESS COMPLEX AND LAST LONGER -REDUCED OVERALL WEIGHT -LOWER COST -FAILURE IS NOT E.O.M.
PERFORMANCE DISADVANTAGES	PERFORMANCE DISADVANTAGES
-HIGH MASS -LARGE FUEL REQUIREMENTS -LEAST RELIABLE -TUMBLES ON FAILURE -SHORTER MISSION LIFE	-DIFFICULT TO ATTAIN COMPARABLE ACCURACIES -SLOWER MANEUVERING -NO EFFECTIVE DELTA-V TRIMS -HIGH POWER REQUIREMENTS -REQUIRES ADDITIONAL SYSTEM FOR WHEEL UNLOADING

Fig. 5.1: 3-AXIS ACTIVE -VS- REACTION WHEEL

REACTION WHEEL MOMENTUM DUMP TRADES

SELECTED SYSTEM THRUSTER UNLOADING ADVANTAGES	BOOM UNLOADING ADVANTAGES
-NO EXTERNAL APPENDAGES -BACK UP CONTROL SYSTEM IN THE EVENT OF WHEEL FAILURE -NOT SENSITIVE TO CONFIGURATION -PLACEMENT IS FLEXIBLE -REDUNDANCY MORE MASS EFFICIENT	-LONGER LIFE (10-20 YEARS) -REDUCED OVERALL WEIGHT -SYSTEM CAN REDUCE GRAVITY EFFECTS -NO MASSIVE FUEL TANKS
DISADVANTAGES	DISADVANTAGES
-MORE MASS -MORE VALVE ACTUATION -FUEL SLOSH CONSIDERATIONS	-STRUCTURAL FLEXING CONSIDERATIONS -CAN GENERALLY UNLOAD ONLY TWO AXES -BOOMS ARE DEPLOYED THROUGHT MISSION

Fig. 5.2: THRUSTER -VS- BOOM UNLOADING

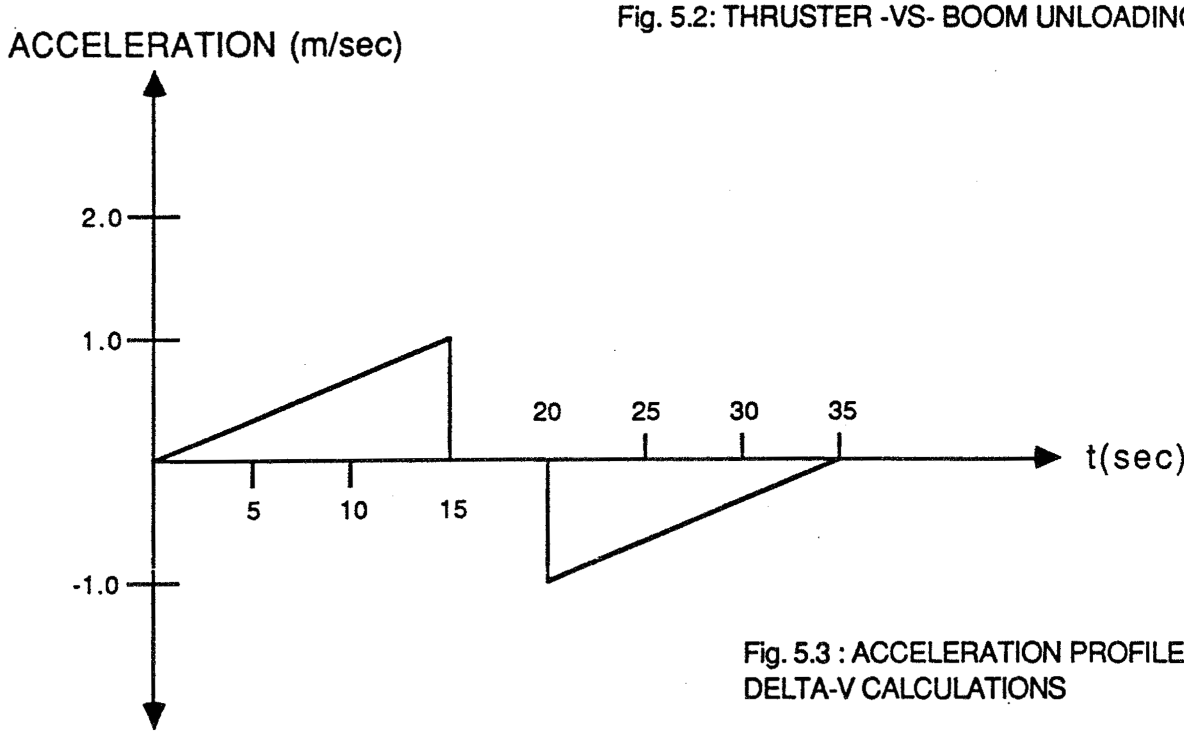

Fig. 5.3 : ACCELERATION PROFILE FOR DELTA-V CALCULATIONS

AACS SATELLITE COMPONENTS

COMPONENT CHOSEN	NO. REQ'D	PEAK POWER (W)	TOTAL WEIGHT (Kg)	PERFORMANCE REMARKS
THRUSTERS	12	--	8.2	NASA STANDARD 1lb. THRUST HYDRAZINE THRUSTER TOTAL IMPULSE=50000 lb/sec : pulses=100000 opperational life of 7 years
LINES AND VALVES	--	8.0	10.0	POWER AND WEIGHT REQUIREMENTS ESTIMATED FROM PAST MISSION DATA
GYROS (FIBER OPTICS ROTATION SENSOR-FORS)	3	10.0	10.0	DRIFT=2X10E-4 deg/hour REQUIRING UPDATE APPROXIMATELY EVERY THREE HOURS NOISE=1X10E-5 deg/sec :MEAN TIME BETWEEN FAILURES 10 YEARS
REACTION WHEELS (BENDIX 1880272)	3	9.9	13.6	SIZE=10.5"DX4.3" : MOMENTUM=2.06ft-lbs-sec AT 900 RPM's : ONE WHEEL ON EACH AXIS
COARSE SUN SENSOR (ADCOLE OAO)	2	0.0	0.2	FOV=180 deg : ACCURACY=2 deg AT NULL SIZE=1.9"x1.9"x1.3"
FINE SUN SENSOR	2	4.4	5.2	FOV=+2,-2 deg EACH AXIS : ACURRACY=5 arc sec SIZE=5"x5"x5"
COMPUTER (TRW CDE-16)	1	19.0	1.8	3K ROM AND 64K RAM : 16 BIT WORDS TO INSURE COMPATIBILITY WITH C^3 SUBSYSTEM RELIABILITY =300K HOURS OPPROXIMATELY
FUEL TANKS	2	4.0	- -	ONE MAIN FUEL TANK REQUIRED AND ONE PRESSURANT TANK
ANTENNA ACTUATOR	1	5.0	7.5	POWER AND WEIGHT VALUES ARE ESTIMATES BASED ON PAST MISSIONS ACTUATION = 0.005 deg/step AND LIFE >4 YEARS
SCIENCE BOOM ACTUATORS	2	20.0	1.5	222 NEWTONS : SIZE 12.7"x6.6"x5.1"

Table 5.1 AACS SATELLITE COMPONENTS

C-8

FINAL TRANSFER VEHICLE DESIGN COMPONENTS

COMPONENT CHOSEN	NO. REQ'D	PEAK POWER (W)	TOTAL WEIGHT (Kg)	PERFORMANCE REMARKS
THRUSTERS	12	--	8.2	NASA STANDARD 5lb. THRUST BIPROPELLANT THRUSTER ;TOTAL IMPULSE=50000 lb/sec : pulses=100000 opperational life of 7 +years
LINES AND VALVES	--	8.0	10.0	POWER AND WEIGHT REQUIREMENTS ESTIMATED FROM PAST MISSION DATA
GYROS (FIBER OPTICS ROTATION SENSOR-FORS)	3	10.0	10.0	DRIFT=2X10E-4 deg/hour REQUIRING UPDATE APPROXIMATELY EVERY THREE HOURS NOISE=1X10E-5 deg/sec :MEAN TIME BETWEEN FAILURES 10 YEARS
COARSE SUN SENSOR (ADCOLE OAO)	2	0.0	0.2	FOV=180 deg : ACCURACY=2 deg AT NULL SIZE=1.9"x1.9"x1.3"
FINE SUN SENSOR	2	4.4	5.2	FOV=+2,-2 deg EACH AXIS : ACURRACY=5 arc sec SIZE=5"x5"x5"
COMPUTER (TRW CDE-16)	1	19.0	1.8	3K ROM AND 64K RAM : 16 BIT WORDS TO INSURE COMPATIBILITY WITH C^3 SUBSYSTEM RELIABILITY =300K HOURS OPPROXIMATELY
THRUSTER BOOM ACTUATORS	4	40	3.0	222 NEWTONS : SIZE 12.7"x6.6"x5.1"

Fig.5.2 transfer vehicle AACS components

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VI. COMMAND, CONTROL & COMMUNICATION

Abstract

In response to the Request for Proposals, the Nifty Orbital Aerobraking Delivery System (NOMADS) was designed to carry a payload consisting of a manned aircraft to the Martian surface within the first part of the next century. To this end, seven spacecraft subsystems were defined to facilitate the design of NOMADS. This chapter deals with the Command, Control and Communication subsystem (C^3) and its main design features: antenna sizing, antenna pointing, and the selection of a complete spacecraft command and control system using mathematical analysis and trade study techniques.

6.0 Introduction

The purpose of this report is to create a Command, Control, and Communication subsystem for NOMADS, a conceptual spacecraft system design required to deliver a manned aircraft to the Martian surface within the first decade of the next century. The communication subsystem will be the only link between the spacecraft, Mars and Earth during all phases of NOMADS mission. Consequently, a thorough analysis of the C^3 design process is needed to ensure a safe and successful trip to Mars. The first step in developing such an analysis is to identify and understand the mission system requirements.

6.1 System Requirements

The system requirements can be broken down into a set of general requirements and a set of specific subsystem requirements (A set of specific requirements exists for each of the spacecraft subsystems). An extended discussion of both sets follow in subsequent sections.

6.1.1 General Requirements

The general system requirements must be satisfied for they are the parameters which set the stage for the whole spacecraft design (They contain the primary objective).

As can be seen from Figure 6-1A, the NOMADS delivery system consists of two primary components: 1.) the payload system which carries the manned Mars aircraft to be dropped to the Martian surface and 2.) the instrument bus which will remain in Martian orbit to serve as a communication satellite for the Mars base and as a link to Earth if so desired. In order to facilitate the design process these two components are broken up into seven integrated subsystems of which C^3 is a part.

Other general requirements equally important to C^3 are: the communication link necessary with the Space Shuttle and Space Station; the use of off-the-shelf hardware and pre-1988 technology; the use of the latest advances in artificial intelligence (relevant to computer choice); and the requirement of design simplicity, reliability, minimum mass and low cost, which eliminates the use of an optical communication link for this Mars mission.

6.1.2 Specific Subsystem Requirements

Directly inherent to the general system requirements are the specific subsystem requirements. They are the responsibilities handled by each specific subsystem. The C^3 subsystem requirements are discussed below and listed in Fig. 6-1B.

C^3 s main concerns are sending and receiving telemetry to and from Earth or to and from Mars. Commands from the DSN or from the Mars base are received by the spacecraft antenna and relayed to the other subsystems. Similarly, data from the spacecraft subsystems can be relayed to the communications subsystem where it can be sent to the respective ground station.

The C^3 subsystem also handles the internal spacecraft communication among the different subsystems. The power switching requirement is an important example of this internal communication, for without it the spacecraft would be powerless!

6.2 System Methodology

A method of achieving these requirements was to establish a design flowchart as seen in Fig. 6-2.

The first step, identifying the mission requirements, has already been discussed in the system requirements section 6.1. The next two steps involved collecting data from primary and secondary resources. The information obtained was then mathematically analyzed, studied and evaluated to determine the three main points of interest: antenna sizing, antenna pointing and design of a command and control system.

Integration of the selected components soon followed; and the question of whether or not an optimal C^3 subsystem design was achieved, determined the final step in the design process. An answer of YES led to a complete and optimal subsystem, while an answer of NO brought the communications engineer back to the mathematical analysis step of the flowchart.

6.3 System Antenna Analysis

The following method was used to determine the spacecraft antenna size:

- Parameters defined
- Equations identified
- Mathematical calculations performed
- Antenna chosen

6.3.1 Parameters

A set of parameters used in finding an optimal antenna design are given in this section. These parameters are a result of system requirements, technological constraints imposed on antenna construction, and typical communication values.

Fig. 6-1A **System Requirements**

Primary Objective: Deliver Payload to Martian Surface

1.) Consists of Two Primary Components:
- Payload System
- Instrument Bus

2.) System Integrated Using Seven Subsytems:
- MMPC
- PPS
- Aerobrake
- Structure
- AACS
- CCC
- Science & Radio Relay

3.) Space Shuttle Delivery to LEO & On-Orbit Assembly at Space Station

4.) Retrievable By Remote Manipulating Device

5.) Design does not Preclude:
- Use on Different Missions
- Carrying Different Payloads

6.) Use Off-the-shelf Hardware

7.) Use Pre-1988 Technology

8.) Design Lifetime of 4 yrs. or Greater

9.) Use Latest Advances in Artificial Technology if:
- Enhances Mission Reliability
- Reduces Mission Cost

10.) Design Should Stress:
- Simplicity
- Reliability
- Minimum Mass
- Low Cost

11.) Four Delivery Systems Built:
- 3 Flight Ready
- 1 Ground Test System

12.) Follow All Mission Science Objectives

R.D.

Fig. 6-1B **C^3 Subsystem Requirements**

1.) Send Telemetry to the Earth DSN and to the Martian ground station.
2.) Collect Telemetry from the Earth DSN and from the Martian ground station.
3.) Relay commands to the other spacecraft subsystems.
4.) Control power switching for the other spacecraft subsystems
5.) Control Outputs and Inputs.

C^3 Subsystem Components

1.) Antenna
2.) On-board Computer
3.) Radio System

R.D.

Method of attack

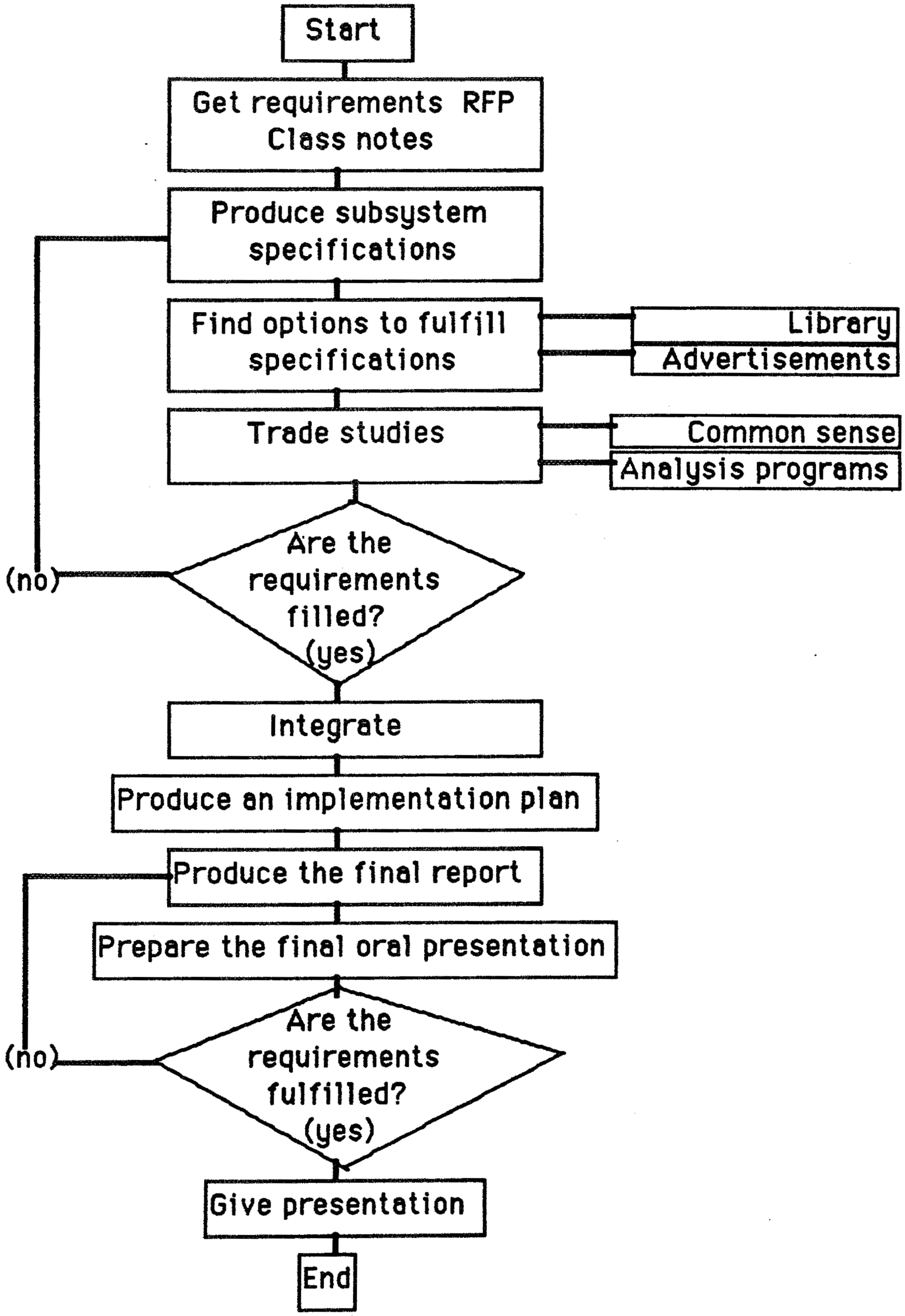


Fig. CDC-1

Parameters cont.

The channels used for deep space communications are [6-5]:

-S band

2.1 GHz Uplink (Earth to S/C)

2.3 GHz Downlink (S/C to Earth)

-X band

7.161 GHz Uplink

8.414 GHz Downlink

For the following analysis, the X band downlink was used. The following fixed parameters were then calculated:

Using X band: wavelength = c/f = 3.57 E-2 m

Using the DSN for S/C to Earth RF link: D_r = 64 m

$A_r = \pi D r^2/4 = 3.2 E3\ m^2$

$G_t = .50(4\pi Ar)/l^2 = 1.7 E7$

$T = 28.5\ ^oK$

z = 50%

6.3.2 Equations Identified

The model for most deep space antenna systems (including the Parabolic antenna) is [6-1]:

$$P_R = \frac{P_T G_T A_{R_{eff}}}{4\pi D^2}$$

It is called the Range Equation, and is most useful to the communication engineer.

A similar form can be constructed by substituting the equations for G_T and A_R above. We then get [6-2]

$$P_r = P_t [(z\pi D_t D_r)/(4 LR)]^2$$

From this equation, the spacecraft diameter, D_t, can be calculated. Using the parameters mentioned in section 6.3.1 and the above equation, the NOMADS high gain antenna was calculated.

Note:

-The maximum Earth to Mars distance was used (R = 2.5 A.U. = 3.74 E11 m) This distance usually varies from 0.5 to 2.5 astronomical units.

-Typical values for P_t and P_r are 20W and 9.5 E-16W respectively [6-1].

$$D_r = \{16(0.0357)^2(3.74\ E11)^2(9.5\ E\text{-}16)\}/\{.50^2(\pi)^2(64)^2(20.0)\}$$

$$\mathbf{D_r = 3.66\ m}$$

Based on the diameter determined above, a high gain antenna with a 3.7m diameter was chosen as seen in Fig. 6-3. Similarly, using the same data above but with different R and P_r values, it was determined that a low gain antenna of diameter of 14 inches would be used for the spacecraft to Mars/spacecraft to manned airplane RF link.

6.3.3 Antenna Pointing and Placement

The placement of the high gain antenna (HGA) will be fixed to the aft of the spacecraft so that it will be pointing toward Earth continuously. The low gain antenna will be placed directly opposite the HGA on the other side of the spacecraft (not only for pointing convenience, but so as not to affect the inertia of this well balanced spacecraft). The mission was broken down into two phases [see Chapter I: Mission Planning]:

6.3.3.1 Phase I:

Launch from L.E.O. & Enroute to Mars

During this phase the HGA will be pointing towards the Earth keeping the communication link open up to phase II and beyond.

6.3.3.2 Phase II:

Aerobrake entry & Mars Final Orbit

During phase II, the HGA will continue to be pointing toward the Earth. In addition, the LGA will begin communications with Mars and the aerobrake reentry system.

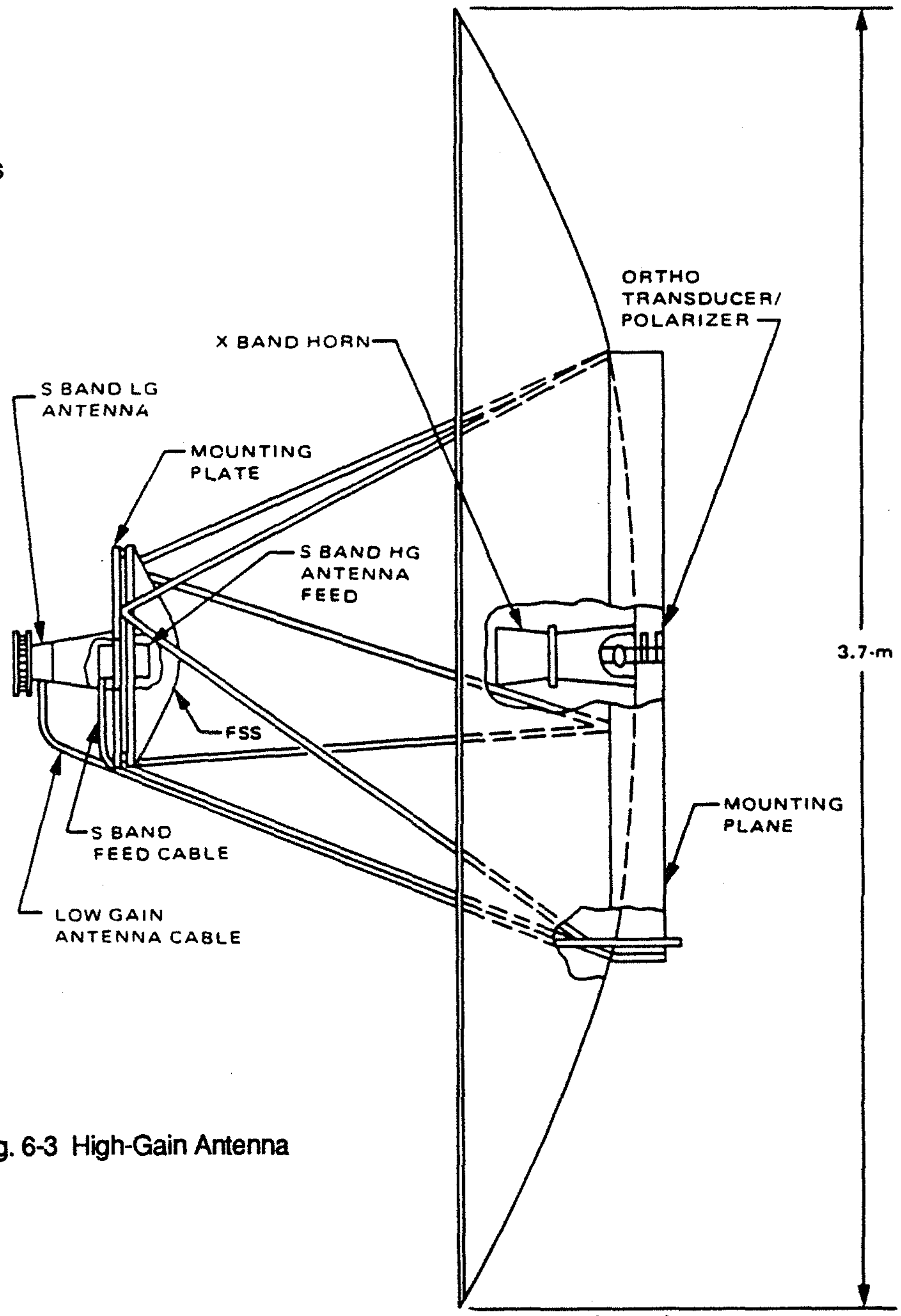


Fig. 6-3 High-Gain Antenna

Just before separation, final commands will be supplied to the Aerobrake system through an interconnecting umbilical chord. After separation, all communications will be conducted via the low gain radio attached to the instrument bus (now a Mars communication satellite).

6.4 System Command and Control Trade Studies:

A system of components was selected on the basis of trade studies and several defined parameters. These parameters are: 1.) the maximum data rate in bits per second (bps) needed to conduct the mission science objectives (See Chapter VII Science Instrumentation Data) and 2.) the geometrical orbit of the satellite around Mars.

Both of these parameters will influence the selection the spacecraft microcomputer, and the need for a large data storage system. Fortunately the maximum instrument data rate is only equal to 2000 bps (again see Chapter VII). Therefore, the need for data storage on board the spacecraft will be greatly reduced. Storage will still be necessary for backup and for the interruptions caused by planetary or solar interference with the telemetry signal. The trade studies conducted for the command and control system are shown below in Table 6-1. An example of a typical command block diagram is shown in Fig. 6-4.

TECHNOLOGY	OPTIONS	CRITERIA	COMMENTS
PROCESSOR-CPU	● MIRCOCOMPUTER ● MINICOMPUTER ● LARGE SCALE	● FAULT TOLERANCE ● COST ● COMPLEXITY ● ARCHITECTURE	● 700-1000 kOPS BY 1995 ● RAD HARD/SPACE QUALIFICATION WILL IMPACT SELECTION
MAIN MEMORY	● CMOS ● MAGNETIC CORE ● PLATED WIRE	● WEIGHT/SIZE /POWER ● COST ● ACCESS TIME ● VOLATILITY	● RAD HARDENING IS A MAJOR FACTOR ● 128k BITS/CHIP BY 1990
MASS MEMORY-ELECTRONIC	● BUBBLE MEMORY ● CCD MEMORY	● WEIGHT/SIZE /POWER ● CAPACITY ● VOLATILITY	● EXPECTED TO ACHIEVE 10^9 BITS/CHIP BY 1990 ● 1.5×10^6 BITS/SEC-TRANSFER RATE
MASS MEMORY-MECHANICAL	● OPTICAL DISCS ● MAGNETIC DISCS ● MAGNETIC TAPE	● STORAGE CAPACITY ● SPACE QUALIFIED	● 2×10^9 BITS/DISC BY 1990 ● 7×10^{10} BITS/TAPE BY 1990
HIGHER ORDER LANGUAGE (HOL)	● ADA ● FORTRAN ● JOVIAL	(SEE HOL TRADE STUDY)	

M84-1086-065PP

Table 6-1 DMS Trade study [6-3]

6.5 System Integration and Optimization

For the final step in the design process, all of the selected C^3 components were integrated and tested. Then, the following question was asked, "Is the integrated system optimal?" If NO, the process would begin anew at the mathematical analysis and trade study level. If YES, the C^3 subsystem design design would be complete. Finally, the optimized C^3 system is integrated with the entire spacecraft to interact and fufill the system requirements. Fig. 6-5 shows these interactions of other subsytems with C^3.

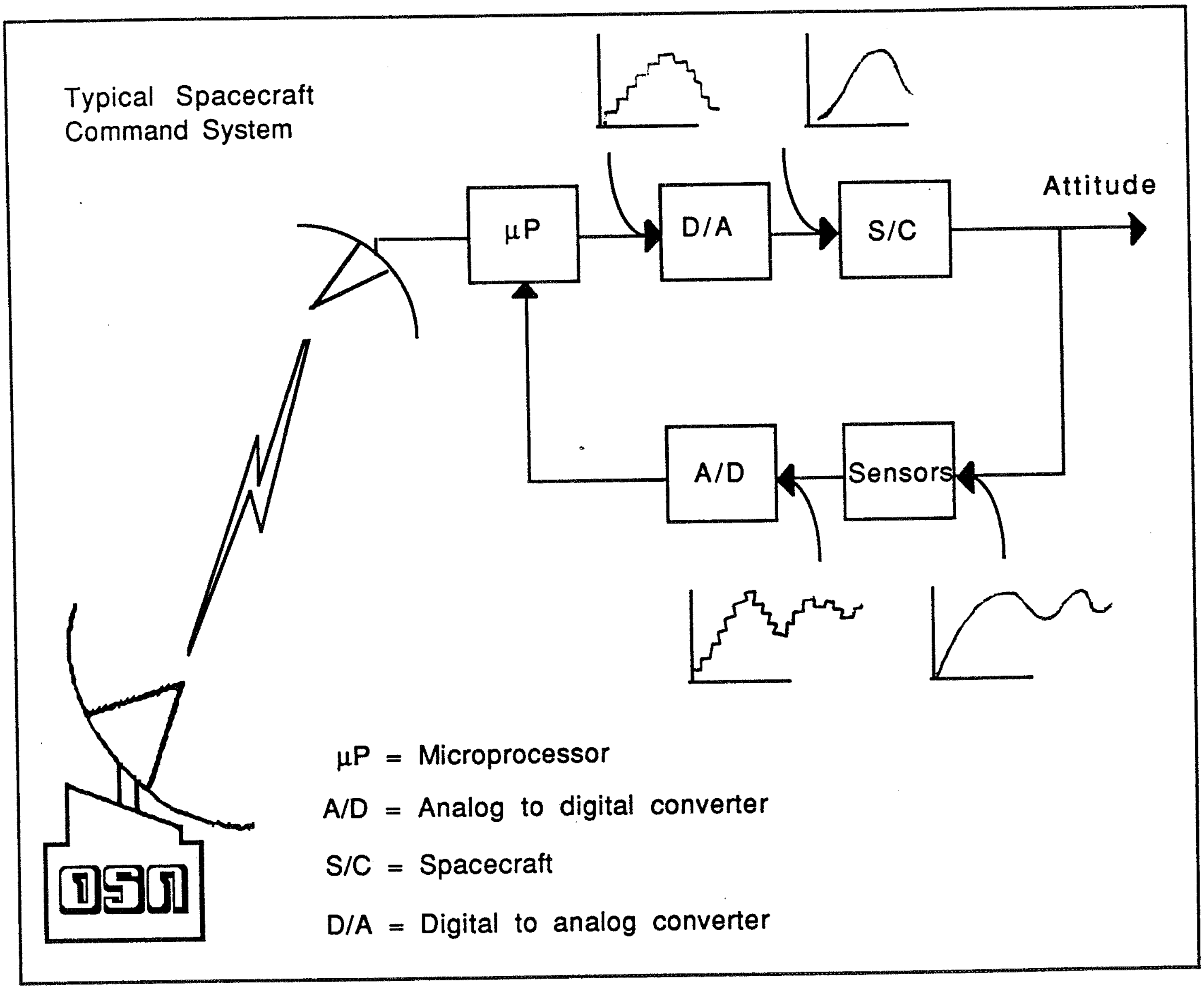
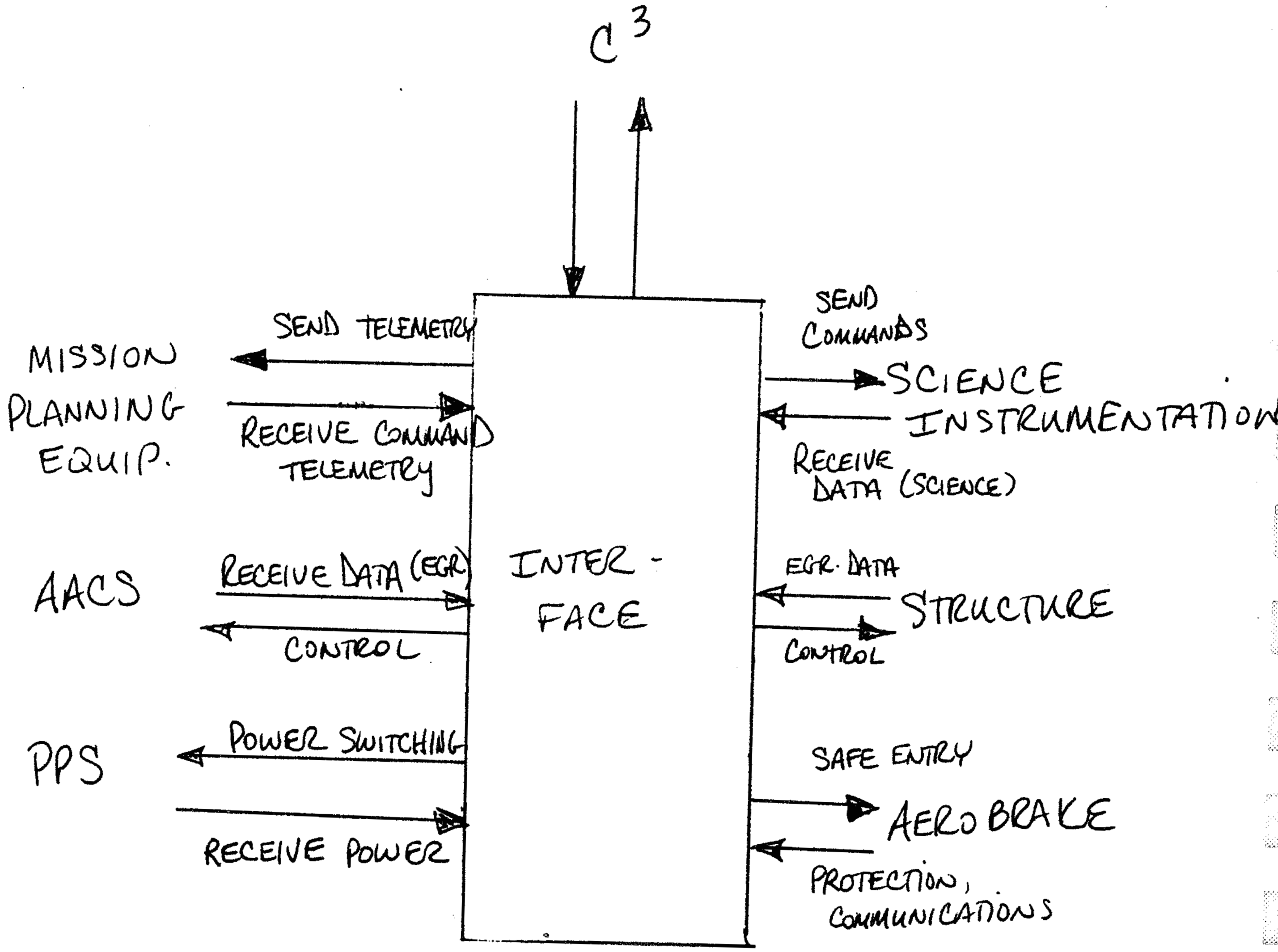


Fig. 6-4 Typical Block Diagram [6-5]



(FIG. 6-5) C^3 INTERACTION WITH OTHER SUBSYSTEMS [6-6]

6.6 Summary of Results (some problems?)

NOMADS completed C^3 subsystem design components:

Antenna Sizes: One HGA: $D_t = 3.7$ m

One LGA: $D_t = 14$ "

Radio: One low frequency radio.

One tape transport (and hardware Fig. 6-6)

Computer & Computer Storage: One microcomputer with CMOS main memory and Bubble memory. (and hardware Fig. 6-6)

*One problem worth noting: The pointing difficulty related to the use of the fixed HGA. Because of the fixed spacecraft position it will on occasion need to be realigned with the DSN. This will only be possible with the realigning of the whole spacecraft using its thrusters. It is therefore probable that a movable antenna might be preferred in spite of the added complexity and cost.

Equipment	Weight (lbs)
1. S-Band TWTA	12
2. S-Band Mod/Driver	3
3. S-Band PLL Receiver	3
4. S-Band Multicoupler	3
5. S-Band Receiver	3
6. Delay Lock Loop Correlator	2
7. UHF Receiver/Demodulator	3
8. Antenna Servo Electronics	8
9. UHF Coupler	0.5
10. Tape Transport	10.5
11. Tape Transport Electronics	10
12. Digital Computer	18
13. Mission Equipment Interface Unit	5.5
14. Spacecraft Interface Unit	4.5
Cables and Connectors	7
Module Base & Covers	36
Subtotal	129
15% contingency	20
Total	149 lbs

Fig. 6-6 CDPI Subsystem Module - Standard Planetary Spacecraft. [6-6]

7.1 Introduction

Although NOMADS' primary mission is to safely transport a Mars airplane to the planet's surface, the stated requirements also include a secondary mission, this being the fulfillment of certain mission science objectives. The instrument bus that travels to Mars as part of NOMADS becomes an orbiting satellite, capable of communicating with the ground/airplane as well as taking scientific measurements of the planet. Certain system requirements have to be fulfilled by the science subsystem as well.

7.2 Science Subsystem Requirements

1. Use off-the-shelf hardware whenever possible.
2. The materials and techniques used must be available before 1998.
3. Instruments should have a lifetime of at least 4 years.
4. Simplicity, reliability, minimum mass and low cost should be stressed.
5. Data collected and formatted to communications subsystem.
6. Power has to be supplied by the power and propulsion subsystem.
7. Fulfill mission science objectives.

7.2.1 Mission Science Objectives

I. Determine the origin, evolution, and present state of the solar system.
II. Obtain a better understanding of the Earth through comparative studies.
III. Understand the relationship between the chemical and physical evolution of the solar system and the appearance of life.

These are further broken down into 6 specific requirements which are then fulfilled through the selection of scientific instruments. These specific requirements are:

1. Determine the elemental and mineralogical character of the Martian surface on a global basis.
2. Determine the distribution, abundance, sources, and concentrations of volatile materials and dust.
3. Define the global gravitational field.
4. Measure the global topography.
5. Explore the atmospheric structure and its circulation in detail.
6. Establish the nature of the Martain magnetic field.

7.3 Selection Of Instruments

The selection of the science instrumentation is a major step in the fulfillment of the subsystem requirements. Not only must the chosen equipment accomplish at least one of the specific requirements, but it also has to meet the system requirements pertaining to lifetime, cost, etc.

The method used for choosing NOMADS' instrumentation was primarily one of library research. Through comparative studies of equipment used on previous space missions, a compilation of eight science instruments was chosen. The most influential of the missions studied was the Mars Geoscience Climatology Observer (7-6), whose mission science objectives are remarkably similar to NOMADS.

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7.3.1 Trade Studies

In order to fulfill the system requirements, instruments from previous missions with a minimum mass and power requirement were selected, whenever possible. Two of the chosen instruments are currently under development for missions with launch dates before that of NOMADS, so they should be off-the-shelf by the time NOMADS is launched. These two were chosen instead of currently available instruments because of the mass and power savings, as well as a simplification of the inertia tensor for the structures subsystem.

One instrument that was not chosen for NOMADS was an imaging system. An imaging system may seem vital for the determination of safe landing sites prior to separation of the instrument bus from the aerobrake/payload, however, imaging systems have extremely high data rates, requiring their own tape backup systems. Also, pointing accuracy must be precise for proper resolution. In order to keep everything as simple as possible, NOMADS' science team decided not to use an imaging system. Instead, the instrumentation for the measurement of global topography will suffice.

7.4 Fulfilling The Requirements

For the determination of the elemental and mineralogical charater of the Martian surface, two instruments were selected. These are a gamma ray spectrometer and a visual and infrared spectrometer. The gamma ray spectrometer will obtain measurements of the elements on the surface, while the visual and infrared spectrometer furnishes a mineralogical map. Previous experience with the gamma ray spectrometer was gained in the Apollo program, whereas the visual and infrared spectrometer is under development for the Galileo mission (7-6).

The Pressure Modulator Infrared Radiometer (PMIRR) will be used to determine the distribution, abundance, sources, and concentrations of volatile materials and dust. This instrument is currently being developed for the Mars Geoscience Climatology Observer (7-5).

The global gravitational field is to be determined using radio science. The radio used is part of the communications subsystem and is discussed in more detail in that section. Radio science experiments have been flown on many NASA missions.

The global topography will be measured using a radar altimeter. A similar experiment was conducted on the Pioneer Venus mission.

A detailed exploration of the Martian atmosphere and its circulation will be carried out by four separate instruments. NOMADS will use radio science, the PMIRR, an ultraviolet spectrometer and an ultraviolet photometer to fulfill this requirement. The PMIRR will obtain atmospheric temperature and water vapor profiles. The ultraviolet spectrometer will determine varying concentrations of ozone in different altitudes, providing more information about the water vapor content. Previous experience with this instrument was gained on missions including Voyager and the Solar Mesosphere Explorer (7-6 and 7-9). The ultraviolet photometer will determine the abundance of atomic hydrogen. This device was flown on Pioneer and Voyager and will be used again on Galileo (7-6, 7-7, 7-9). Radio science will measure the reflectivity of the atmosphere.

In order to etablish the nature of the Martian magnetic field, a magnetometer will be included on NOMADS. Magnetometers have been flown on many planetary spacecraft

(7-6).

In choosing these eight instruments, the mission science objectives are fulfilled (table 1).

7.5 Instrument Placement

All of NOMADS' science instruments will be attached directly to the satellite and will face the planet's surface at all times (fig. 1). A scan platform was deemed unnecessary because of the oscillating mirrors found in the scanning instruments. Also, the planned orbit of the satellite allows for full coverage of the planet without movement of the instruments, yet still makes full use of their fields of view (fig. 2). The microprocessor supplied by the communications subsystem has the capablity of handling data from all of the instrument at once, allowing for flexibility in the planning of experimental modes.

Special attention had to be taken when mounting the magnetometer and the gamma ray spectrometer since the magnetic and electrical fields generated by the satellite and other instruments interfere with their sensing. Therefore, both of these instruments are mounted on opposing booms on the satellite.

7.6 Interaction With Subsystems

In order for the satellite and its instruments to function properly, the science subsystem has to interface with the six other subsystems.

The mission planning subsystem needs to know the total mass of the instruments (table 2) for determining the delta v needed to reach Mars as well as a viable orbit for the instrument bus. Mission planning also needs to know the orbit requirements of the instruments to plan an orbit that will benefit all of the subsystems. The science subsystem requires an orbit that will allow enough time to carry out the planned experiments, as well as be within the range of the instruments' capabilities.

The aerobrake subsystem also needs to know the total mass for determining the size of the shield needed.

The power and propulsion subsystem needs to know the power requirements of the instruments in order to size the solar panels for the instrument bus (table 2).

The communications subsystem requires the data rates in order to select a microprocessor capable of handling the flow of data (table 2). The communications subsystem for NOMADS also needed to be informed of the radio science experiment so as to properly place the radio equipment.

The structures subsystem needed the instrument placement requirements as well as the mass and dimensions of the instruments in order to properly balance the satellite (table 2).

The attitude and articulation control subsystem (AACS) needed the pointing requirements for the various instruments. The AACS for NOMADS primarily needs to maintain the position of the instrument bus in orbit, a job greatly simplified with the deletion of the scan platform.

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7.7 Problems

The main problem encountered by the science instrumentation subsystem is a lack of data pertaining to the expected lifetime of the instruments. An assumption had to be made that the instruments previously used on other missions had to have lasted at least until their purposes were accomplished and that those under develpoment would also last on their respective missons until their purposes were accomplished. This leaves NOMADS science instruments with an expected lifetime of a few years.

7.8 References

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Reqmnt / Instr	Elemental and mineralogical character	Volatile materials and dust	Global gravitational field	Global topography	Atmospheric structure and circulation	Martian magnetic field
Gamma Ray Spect.	X					
UV-vis-IR Spect.	X				X	
Radar Altimeter				X		
PMIRR		X			X	
UV Photometer					X	
Radio Science			X		X	
Magnetometer						X

Table 1

Data / Instr	MASS (kg)	POWER (watts)	DATA RATE (bps)	DIMENSIONS
Gamma Ray Spect.	14.0	5.0	2000	40x40x30
UV-vis-IR Spect.	6.0	12.0	200	31x40x40
Radar Altimeter	7.0	17.0	625	1-m dish + 29.4x29.4 x19.6
PMIRR	25.7	27.1	150	81x33.5x24 + 70x40x63
UV Photometer	0.6	1.0	?	10.1x10.1 x12.7
Radio Science	5.0	18.0	N/A	20.3x63.5 x12.7
Magnetometer	6.0	6.0	150	20.3x12.7 x10.1
TOTALS	64.3	86.1	3125+	

Table 2

INSTRUMENT PLACEMENT

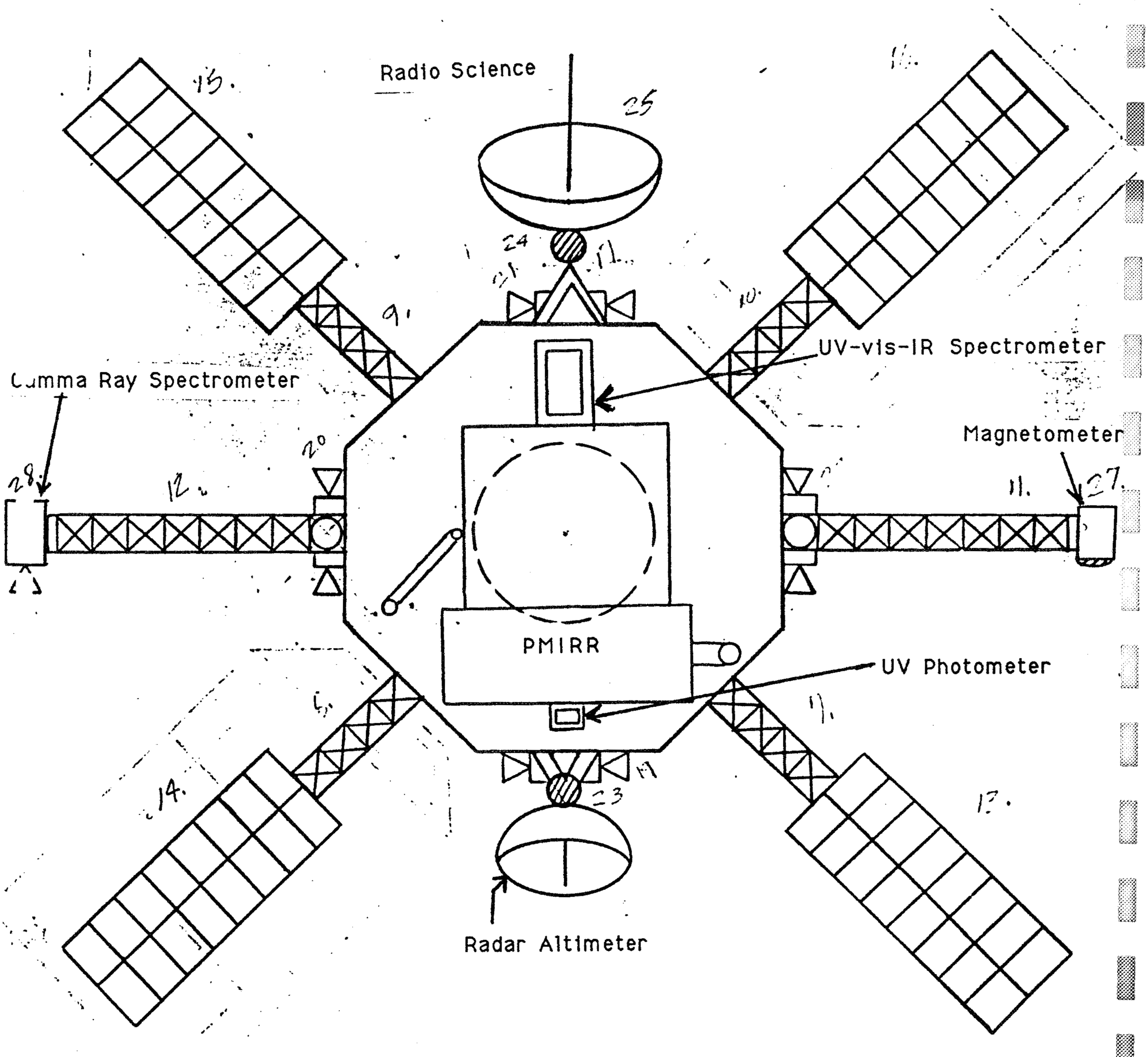

Fig. 1

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FIELDS OF VIEW

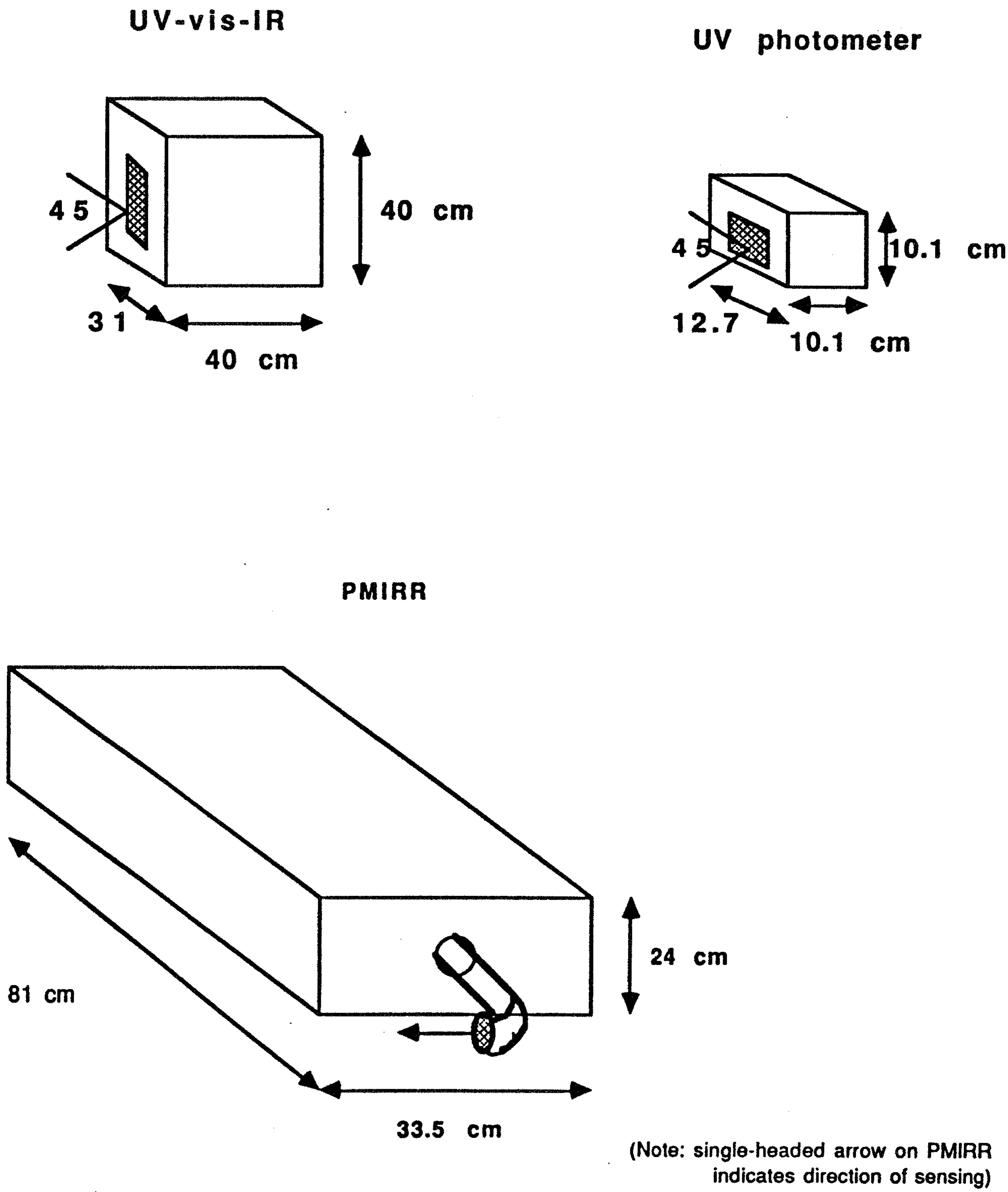

Fig. 2

= sensing face

(7.8)

Appendix A. spacecraft subsystem mass estimates

subsystem	mass (kg)
structure	3683.0
science	59.0
aerobrake	939.6
AACS	458.5
C3	90.0
power	102.4
propulsion	21,289.0
payload	1021.0

TOTAL MASS OF SPACECRAFT 27,650.0 KG

NOTE: Propellant tank masses are included in the structure subsystem.